

**BOEING**

**GENERAL  
ELECTRIC**

CR-151673  
D180-24071-3

NAS9-15196  
DRL T-1346  
DRD MA-665T  
LINE ITEM 4

**Part III**  
**Final Briefing**  
**March 7, 1978**

N78-20164

(NASA-CR-151673) SOLAR POWER SATELLITE  
SYSTEM DEFINITION STUDY. PART 3: Final  
Briefing (Boeing Aerospace Co., Seattle,  
Wash.) 447 P HC A19/ME A01 CSCI

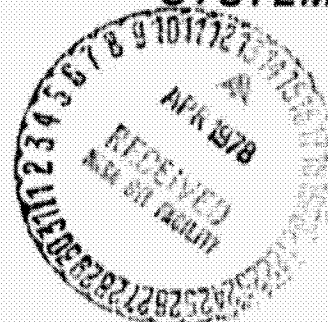
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# Solar Power Satellite

**SYSTEM DEFINITION STUDY**  
**PART III**



D180-24071-3

# Solar Power Satellite

## *SYSTEM DEFINITION STUDY PART III*

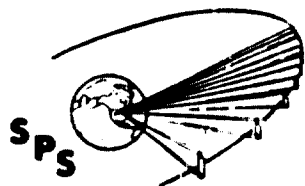
**Part III  
Final Briefing  
March 7, 1978**

Approved:



**G. R. Woodcock  
Study Manager**

Boeing Aerospace Company  
Missiles and Space Group  
Space Division  
P.O. Box 3999  
Seattle, Washington 98124



SPS-2000

**BOEING**

# **Solar Power Satellite Systems Definition Study Part III**

## **Final Briefing**

### **AGENDA**

**INTRODUCTION AND HIGHLIGHTS** \_\_\_\_\_ **G. WOODCOCK**

**SYSTEM DESCRIPTION** \_\_\_\_\_ **G. WOODCOCK**

**SIZE SENSITIVITY** \_\_\_\_\_ **G. WOODCOCK**

**TRANSPORTATION** \_\_\_\_\_ **H. DIRAMIO**

### **EMPHASIS AREAS**

**OPERATIONS AND MAINTENANCE** \_\_\_\_\_ **E. DAVIS**

**MPTS STUDIES** \_\_\_\_\_ **E. NALOS**

**POWER GRID INTERFACE** \_\_\_\_\_ **BJORN KAUPANG**

**DEVELOPMENT REQUIREMENTS FOR  
INITIAL SPS COMMERCIALIZATION** \_\_\_\_\_ **D. GREGORY**

**D180-24071-3**

### **PART 3 HIGHLIGHTS**

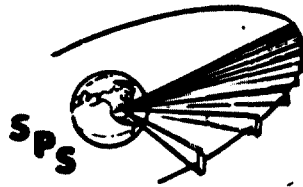
The principal design change made in the reference system was to modify the transmitter design for maintainability and crew access.

At the completion of Part II, the comparison between ballistic and winged launch vehicles indicated difficulties with the winged vehicle: inadequate payload bay volume and requirement for downrange recovery of the booster. A winged vehicle has now been configured with larger payload volume and with a flyback booster. Its cost per flight is competitive with the ballistic sea landing vehicle.

A preliminary maintenance analysis was completed. The estimated cost for manned SPS maintenance (every six months) is about 3% of the power cost attributable to initial system capital cost: approximately 1 mill per kilowatt hour.

The far sidelobes study for SPS transmitters showed that the grating lobes are widely separated points rather than rings. This minimizes concern with overlap of grating lobes. The levels of grating lobes can be reduced by improved mechanical pointing. The dominant contributor to grating lobe magnitude is the saw-toothing of the wave front from the subarray that results from mechanical aiming errors.

Requirements for SPS demonstration are not clear at the present time. It is possible that development might include a technical demonstration of economic viability. This would be likely in scenarios including a commercial funding contribution to the development program. Technical demonstration options were evaluated and a preferred demonstration approach selected.



SPS-1000

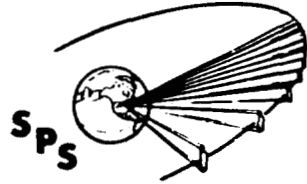
## Part III Highlights

**BOEING**

- TRANSMITTER DESIGN MODIFIED FOR MAINTAINABILITY
- TWO-STAGE WINGED LAUNCH VEHICLE COMPETITIVE WITH BALLISTIC OPTION
- MAINTENANCE COSTS ADD ~3% TO POWER COST
- TRANSMITTER GRATING LOBES ARE WIDELY SEPARATED POINTS  
—LEVELS CAN BE REDUCED BY IMPROVED MECHANICAL POINTING
- SPS DEMONSTRATION OPTIONS EVALUATED

### **SYSTEM SELECTION RATIONALE**

A summary of the rationale for system selection is presented on the facing page. This rationale has been aimed at maximizing credibility of the SPS concept and at improving confidence in mass cost and technology estimates. If the resulting system had been too massive or too costly then it would have been necessary to step forward to more advanced technology as a reference design. However, the results we have found indicate that the system selected is adequate in terms of mass, performance, and cost.



SPS-1999

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## System Selection Rationale

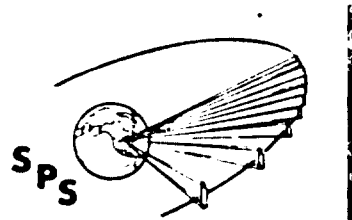
**BOEING**

- CREDIBILITY OF THE SPS SYSTEM CONCEPT IS MAXIMIZED BY MINIMIZING TECHNOLOGY EXTRAPOLATIONS
- SYSTEM DEFINITION UNCERTAINTY IS MINIMIZED BY MINIMIZING TECHNOLOGY EXTRAPOLATION
- THE RESULT IS AN ADEQUATE SYSTEM
- MORE ADVANCED TECHNOLOGY WILL EVENTUALLY YIELD BETTER SYSTEMS.



**SELECTION EXAMPLE**

Three examples of the selections among various options are indicated here. Single crystal silicon was selected over thin film gallium arsenide because of its greater technology and production base and better overall understanding. The transmitter selection of klystrons over amplitrans was comparatively arbitrary and principally motivated in order to develop design detail on the klystron option. The launch system was a comparatively conservative technology two-stage reusable rocket similar in many respects to the fully reusable shuttle concepts examined in 1970 and 1971 except for its larger size.



D180-24071-3

## Selection Examples

SPS-1998

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- **SOLAR CELL OPTIONS:**

**SINGLE-CRYSTAL SILICON**

THIN-FILM GALLIUM ARSENIDE

OTHER THIN FILMS

- **TRANSMITTER OPTIONS:**

**KLYSTRON**

AMPLITRON

SOLID-STATE

- **LAUNCH SYSTEM OPTIONS:**

**TWO-STAGE ROCKET**

SSTO ROCKET

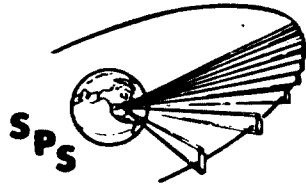
AIRBREATHER/ROCKET MIXES

LASER AND OTHER ADVANCED PROPULSION

**D180-24071-3**

### **SINGLE CRYSTAL SILICON**

Although single crystal silicon has been regarded as a low technology selection, a serious examination of technology advancement requirements indicate that several difficult tasks are involved in attaining the needed performance and production levels. It is quite likely that these tasks represent the schedule tent pole for an SPS program.



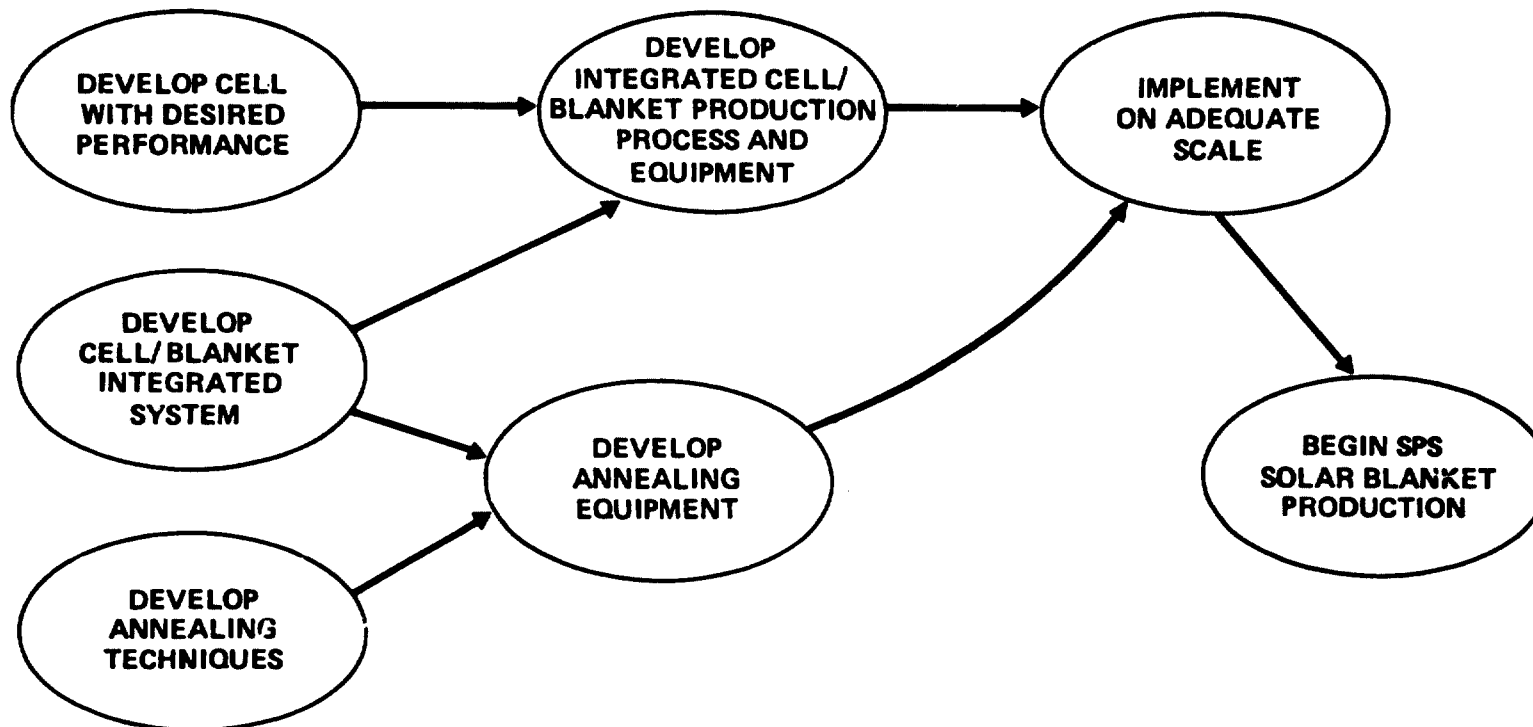
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## Single-Crystal Silicon

SPS-1987

BOEING

SINGLE-CRYSTAL SILICON IS OFTEN VIEWED AS A "LOW TECHNOLOGY" APPROACH TO SPS.....



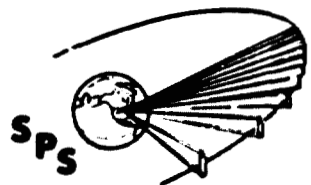
BUT THIS APPEARS TO BE THE SCHEDULE TENT POLE FOR AN SPS PROGRAM.

**D180-24071-3**

**THERMALLY ANNEALED SILICON SOLAR CELLS**

The directed energy annealing effort was continued into Part III and the principal result was a successful anneal of solar cell using a laser. Illustration of the technique and the performance achieved are shown on the chart.

# Thermally Annealed Silicon Solar Cells



SPS-1084

 $2 \times 10^{15}$  ONE-MeV ELECTRONS

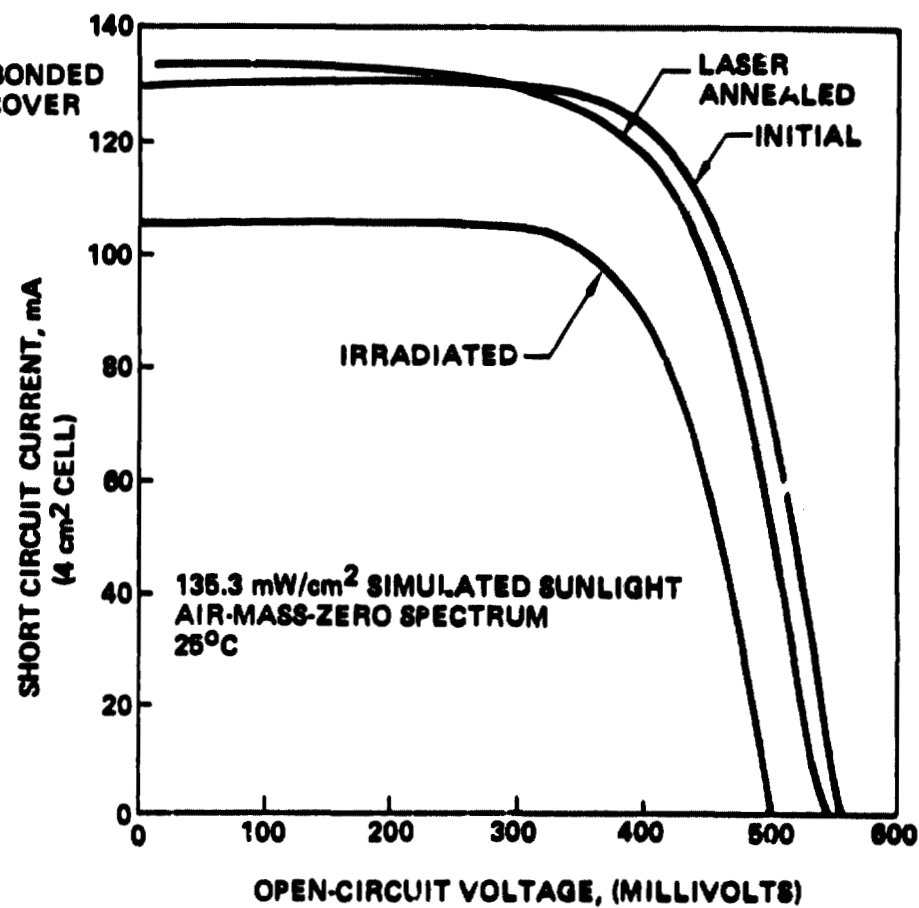
 $\text{Ta}_2\text{O}_5$  ANTI-REFLECTION COATING

ELECTROSTATICALLY BONDED BOROSILICATE GLASS COVER

CO<sub>2</sub> LASER

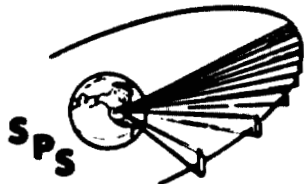
CELL MOTION

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**ROUTES TO HIGH EFFICIENCY IN SILICON SOLAR CELLS**

An approach to achieving the desired cell performance is reasonably clear although the elements of the technology have never been combined in a single device, and one of the elements has not been demonstrated experimentally. Current 50 micrometer single crystal silicon cells are nearing 12% efficiency. Thicker cells are in the 14-15½ range. A combination of the COMSAT non-reflective sculpturing technique with the hi-lo junction emitter technique (and possibly back surface field), combined with diffractive saw-tooth glass covers, is predicted to yield a 17% efficient 50 micrometer solar cell.

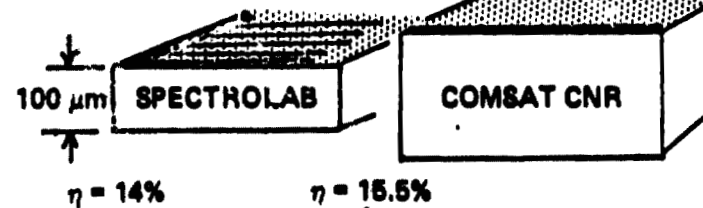
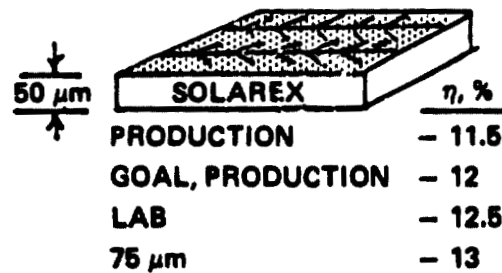


SPS-1085

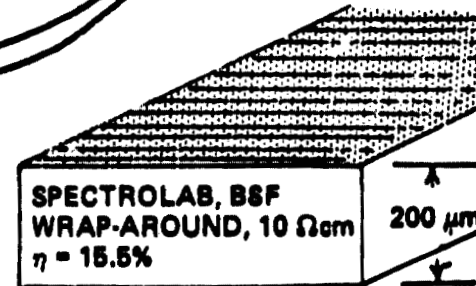
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# Routes to High Efficiency in Silicon Solar Cells

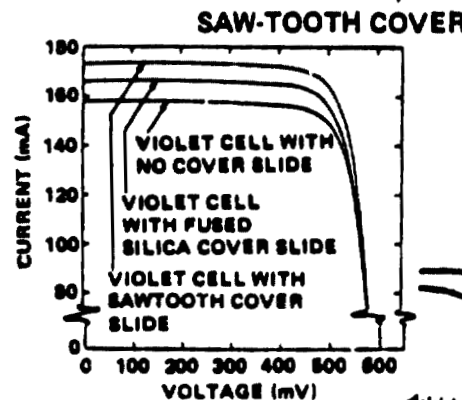
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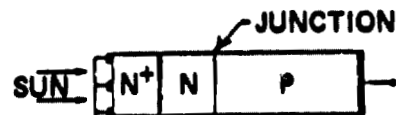
COMSAT CNR



$V_{oc}$  = 0.595V  
 $I_{sc}$  = 41.75 mA/cm<sup>2</sup>  
FF = 0.83



**HIGH-LOW JUNCTION  
EMITTER STRUCTURE**



TODAY  
HLE

|       | A/cm <sup>2</sup> | $V_{oc}$ | FF    | $\eta$ |
|-------|-------------------|----------|-------|--------|
| TODAY | 0.028             | 0.590    | 0.817 | 14.5%  |
| HLE   | 0.031             | 0.626    | 0.804 | 16.9%  |

(PER SAH, LINDHOLDM, FOSSUM)

50  $\mu\text{m}$



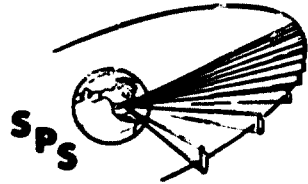
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### **SPS WORK BREAKDOWN STRUCTURE**

The SPS system description follows this work breakdown structure. The structure is basically similar to that used in Part 2 with the exception of adding maintenance items under the SPS Space Construction block. This briefing shows only significant changes to the system description included in the final briefing and final documentation of Part 2. The system description document, to be provided as a part of the Part 3 documentation, will present a formal system description in its entirety.

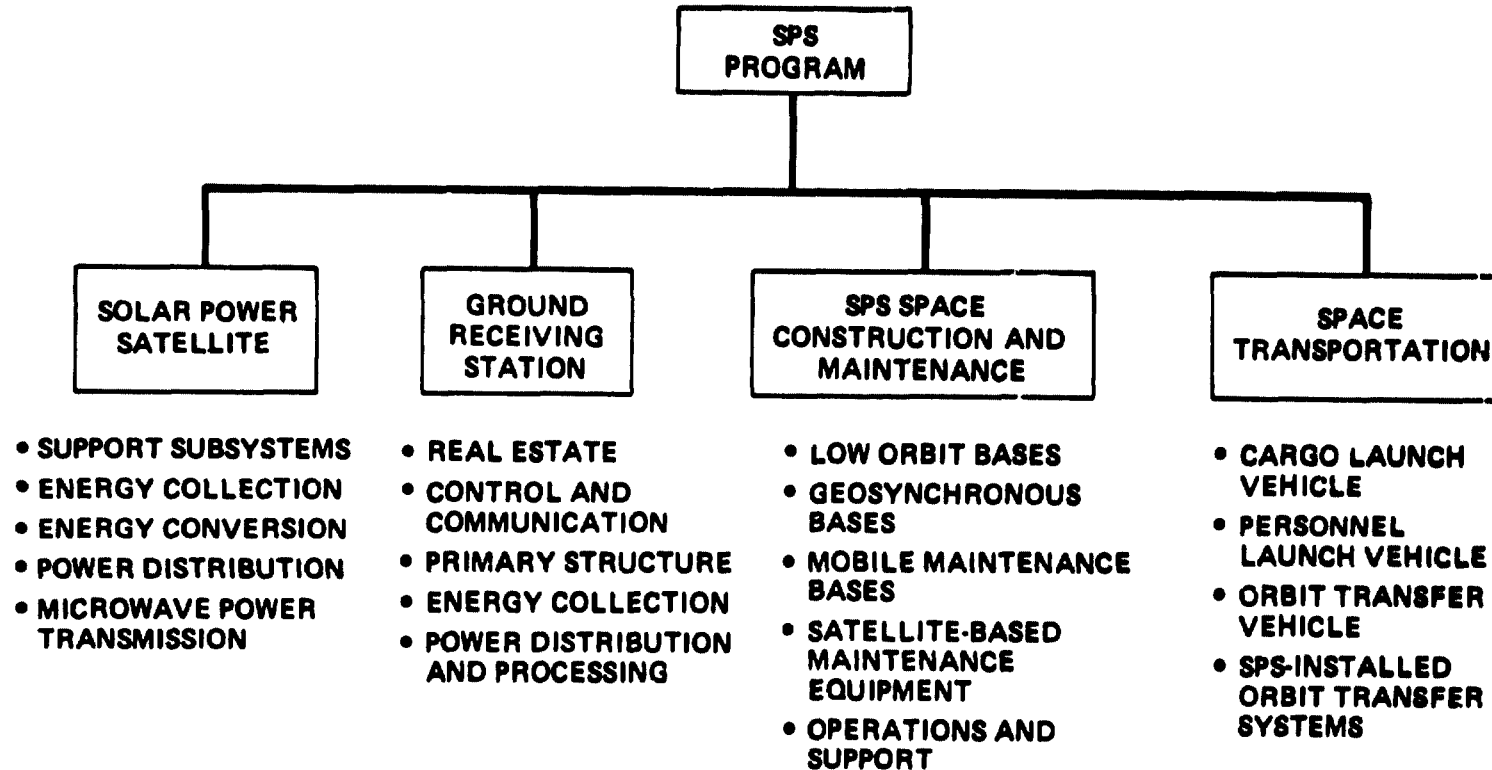
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SPS-1996

# SPS Work Breakdown Structure

*BOEING*



### REFERENCE SYSTEM POWER BUDGET AND SIZING CRITERIA

These are the factors used in calculating the solar array power output. We start with solar cells having 15.75 percent efficiency. To this we add a 10 percent improvement, which could be achieved by any one of several means. For example, A. Meulenberg of COMSAT Laboratories estimates that the sawtooth cover that he invented will improve the efficiency of solar cells by 8 to 12 percent.

The blanket factors of 0.9453 account for the power losses shown. The individual elements of the blanket factors will change, but the product will probably remain around 0.9453.

The summer solstice loss accounts for the 23.5 degrees mis-orientation with respect to the Sun's rays. This loss could be avoided by having the satellite oriented perpendicular to the ecliptic plane, but the cost in thrusters and propellants required for attitude control in that mode shows to no real advantage.

The aphelion intensity factor accounts for the reduced solar intensity when the Earth is at its aphelion, around the first part of July.

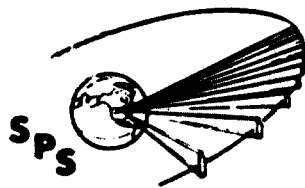
The temperature losses result from the solar cells operating between 36.5°C and 46°C, rather than at the 25°C at which cell efficiency is commonly tested.

The output is further reduced by 3 percent to account for radiation damage that cannot be removed by thermal annealing. In past tests, 95 percent of the radiation damage in solar cells has been annealed out, even though the cells had not been designed for thermal annealing. There is no theoretical reason why all of the radiation damage in solar cells cannot be annealed out, annealing temperatures of around 500°C being well below the 800°C region where diffusion of impurities starts. On the other hand, the operating plan for the solar power satellite involves repeated annealings, which have not been attempted by anyone, as far as we know.

Those sections of the solar array blanket that are used for the orbit transfer power supply, are subjected to a significantly higher radiation degradation than the stowed solar array. If, after annealing, the highly degraded portions of solar array can only be restored to 95 percent of their initial output, a penalty results. This is compensated for by an orbit transfer power compensation factor and by increasing the solar cell string lengths.

The array power requirement of  $18.31 \times 10^9$  watts is based on providing a ground output of  $10.0 \times 10^9$  watts using the current efficiency chain from the array to the grid interface.

The array power requirement and the effective blanket output determine the solar cell area requirement. The increase shown for the array area compensates for the lost areas in the solar array blanket. Another area increase, to compensate for non-array lost area, is necessary to establish the total projected satellite area.



# Reference System Power Budget and Sizing Criteria

SPS-1847

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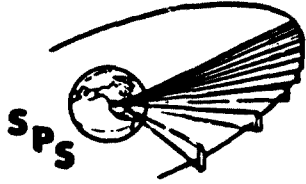
|                                                                               |          |                                 |
|-------------------------------------------------------------------------------|----------|---------------------------------|
| • EFFECTIVE BLANKET OUTPUT—180.6 w/m <sup>2</sup>                             | (E.O.L.) |                                 |
| • BASIC CELL PERFORMANCE (0.1575 @ AMO-25°C)                                  |          | — 213.1 w/m <sup>2</sup>        |
| • 10% IMPROVED PERFORMANCE—DUE TO TEXTURED COVERS                             |          | — 234.4 w/m <sup>2</sup>        |
| • BLANKET FACTORS—STRING I <sup>2</sup> R, UV LOSSES, & MISMATCH (0.9453)     |          | — 221.6 w/m <sup>2</sup>        |
| • TEMPERATURE LOSSES—36.5°C @ SUMMER SOLSTICE (0.9540)                        |          | — 211.4 w/m <sup>2</sup>        |
| • SUMMER SOLSTICE COSINE, LOSSES (0.9190)                                     |          | — 194.3 w/m <sup>2</sup>        |
| • APHELION INTENSITY FACTOR (0.9675)                                          |          | — 188 w/m <sup>2</sup>          |
| • 30-YEAR NON-ANNEALABLE RADIATION DEGRADATION (0.970)                        |          | — 182.3 w/m <sup>2</sup>        |
| • ORBIT TRANSFER COMPENSATION (.9906)                                         |          | — 180.6 w/m <sup>2</sup>        |
| • ARRAY POWER REQUIREMENT—18.31 (10) <sup>9</sup> WATTS                       |          |                                 |
| • GROUND OUTPUT                                                               |          | — 10.0 (10) <sup>9</sup> WATTS  |
| • SLIP RING TO GROUND OUTPUT EFFICIENCY LINK (1.693)                          |          | — 16.93 (10) <sup>9</sup> WATTS |
| • SATELLITE BUS I <sup>2</sup> R LOSSES (1.071)                               |          | — 18.13 (10) <sup>9</sup> WATTS |
| • OVERSIZE—REGULATION, AUX. PWR., ANNEALING (1.01)                            |          | — 18.31 (10) <sup>9</sup> WATTS |
| • SOLAR CELL AREA REQUIREMENT—101.4 km <sup>2</sup>                           |          |                                 |
| • ARRAY AREA REQUIREMENT (INCLUDES LOST AREAS ON ARRAY)—110.2 km <sup>2</sup> |          |                                 |
| • TOTAL SATELLITE AREA (EXCLUDING ANTENNAS)—114.5 km <sup>2</sup>             |          |                                 |

**REFERENCE PHOTOVOLTAIC SYSTEM DESCRIPTION**

The Part II silicon photovoltaic system provided an output of 4650 megawatts per antenna. To normalize this output to 5000 megawatts it was necessary to increase the satellite bay size to 667.5 meters which was more than adequate to satisfy the increased area requirement.

Shown here is the final reference system size and configuration. Details are shown of a typical bay and the array support within the bay.

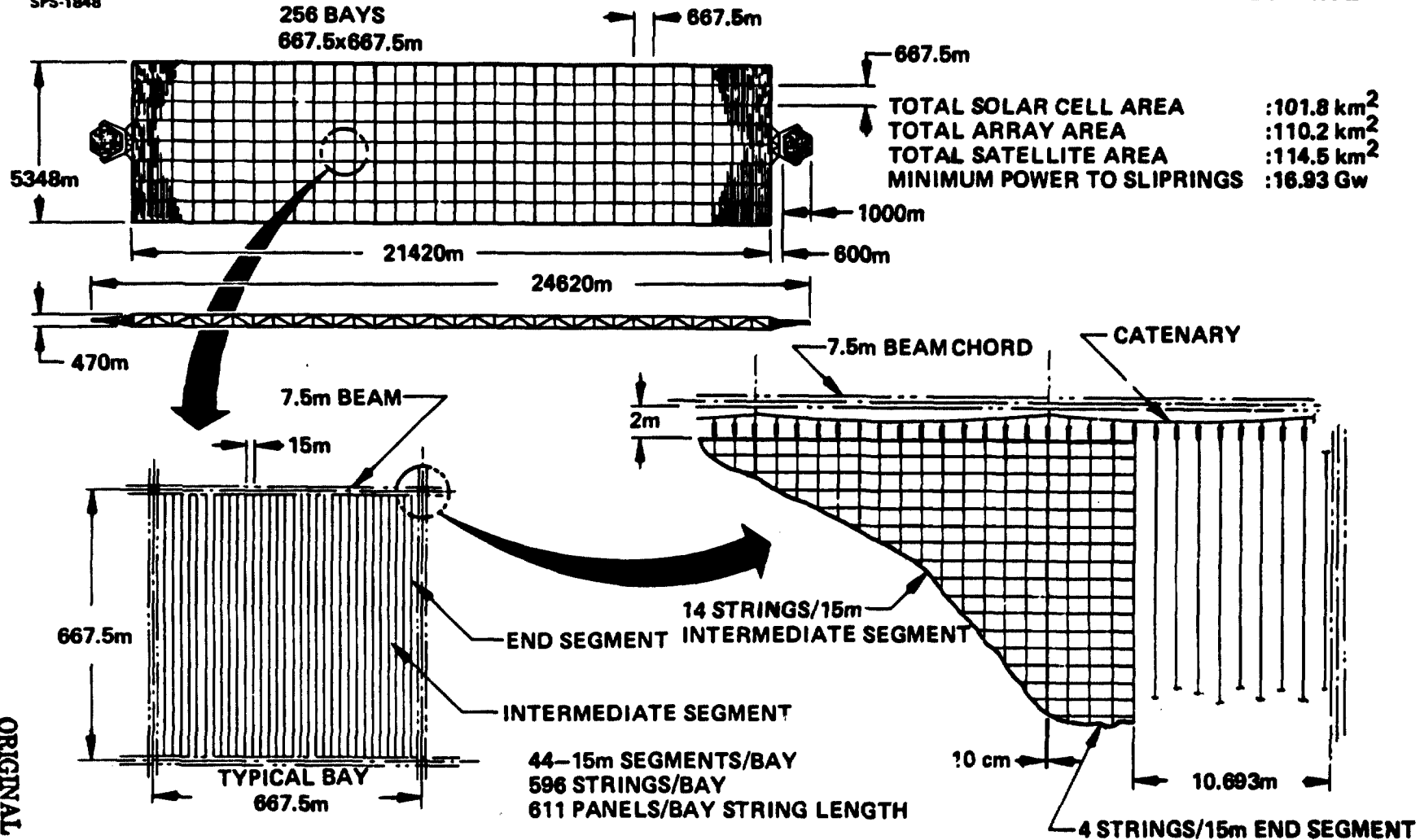
The array segment width was changed to 14.9 meters. This change provided better packaging for transport but made it necessary to provide 15 meter catenary attachment points on the structural beams. A 10 cm spacing was provided between array segments for clearance during array deployment.



# Reference Photovoltaic System Description

SPS-1848

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**REFERENCE PHOTOVOLTAIC SYSTEM DESCRIPTION**

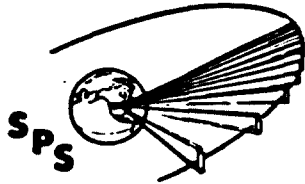
This is the basic panel adopted for design studies. It has a matrix of 224 solar cells, each 6.55 by 7.44 cm in size, connected in groups of 14 cells in parallel by 16 cells in series. The cells are electrostatically bonded between two sheets of borosilicate glass. Spacing between cell and edge spacings are as shown. Tabs are brought out at two edges of the panel for electrically connecting panels in series. Cells within the panel are interconnected by conducting elements printed on the glass substrate.

Important panel requirements were these:

- o The panel components and processes should be compatible with thermal annealing at 500°C.
- o Presence of charge-exchange plasma during ion-engine operation may necessitate insulating the electrical conductors on the panel.
- o The panel design should be appropriate for the high-speed automatic assembly required for making the some 93 million panels required for each satellite.
- o Low weight and low cost are important.

Also shown here is the way panels would be assembled to form larger elements of the solar array. The interconnecting tabs of one panel are welded to the tabs of the next panel in the string, and then the interconnections are covered with a tape that also carries structural tension between panels. After joining, the panels are accordion-folded into a compact package for transport to the low-Earth-orbit assembly station.

The 0.5 cm spacing between panels provides room for the welding electrodes, and also permits reasonable tolerances in the large sheet of 75  $\mu\text{m}$  glass that covers the cells and the 50  $\mu\text{m}$  sheets of substrate glass.

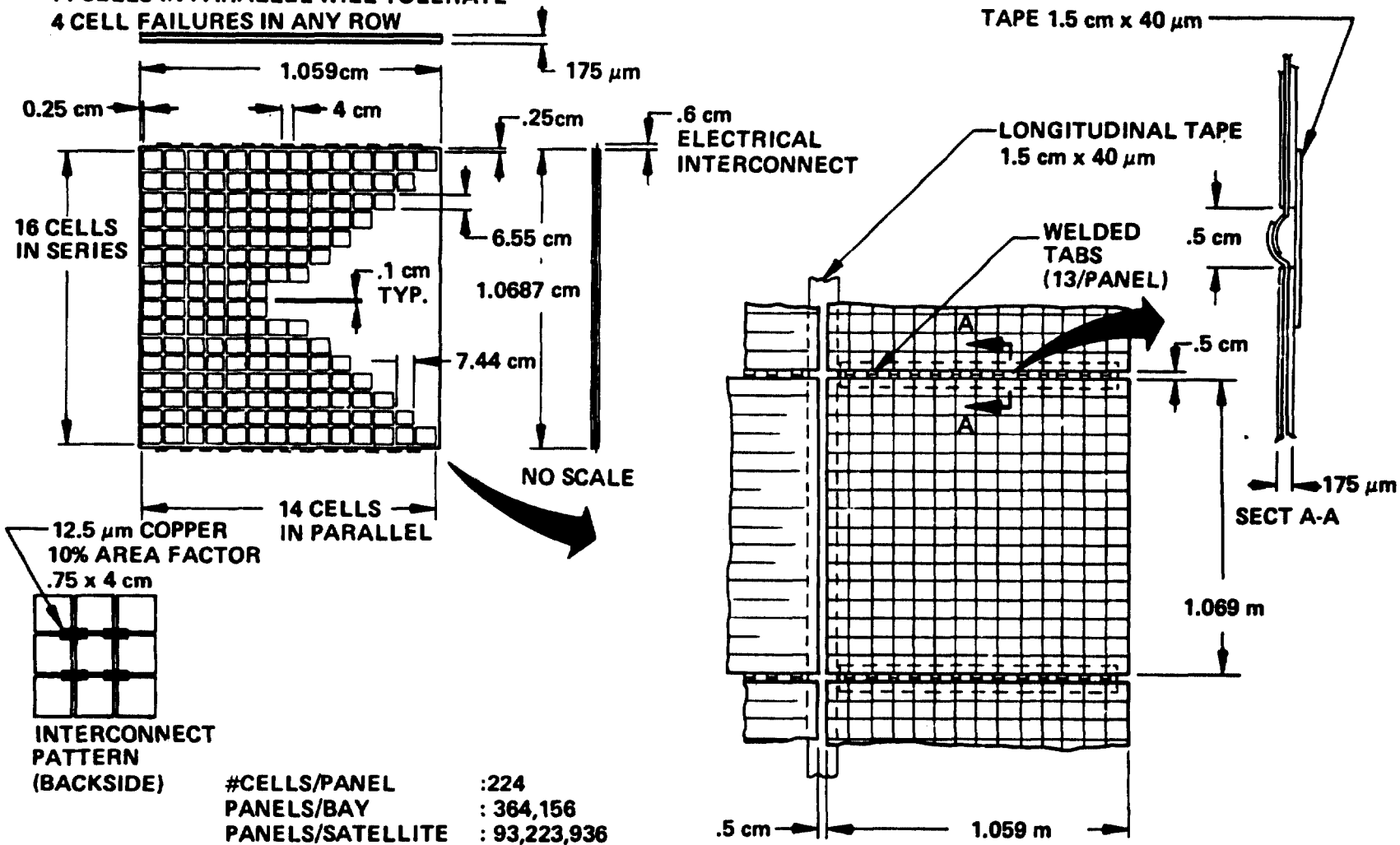


# Reference Photovoltaic System Description

SPS-1849

- 14 CELLS IN PARALLEL WILL TOLERATE 4 CELL FAILURES IN ANY ROW

BOEING



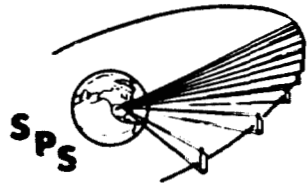


### **REFERENCE ARRAY BLANKET SUPPORT**

This illustration shows the method of providing tension to the solar array blanket segments. This method of support will provide a uniform tension to the end of each array segment by the use of constant-force compression springs at each blanket support tape.

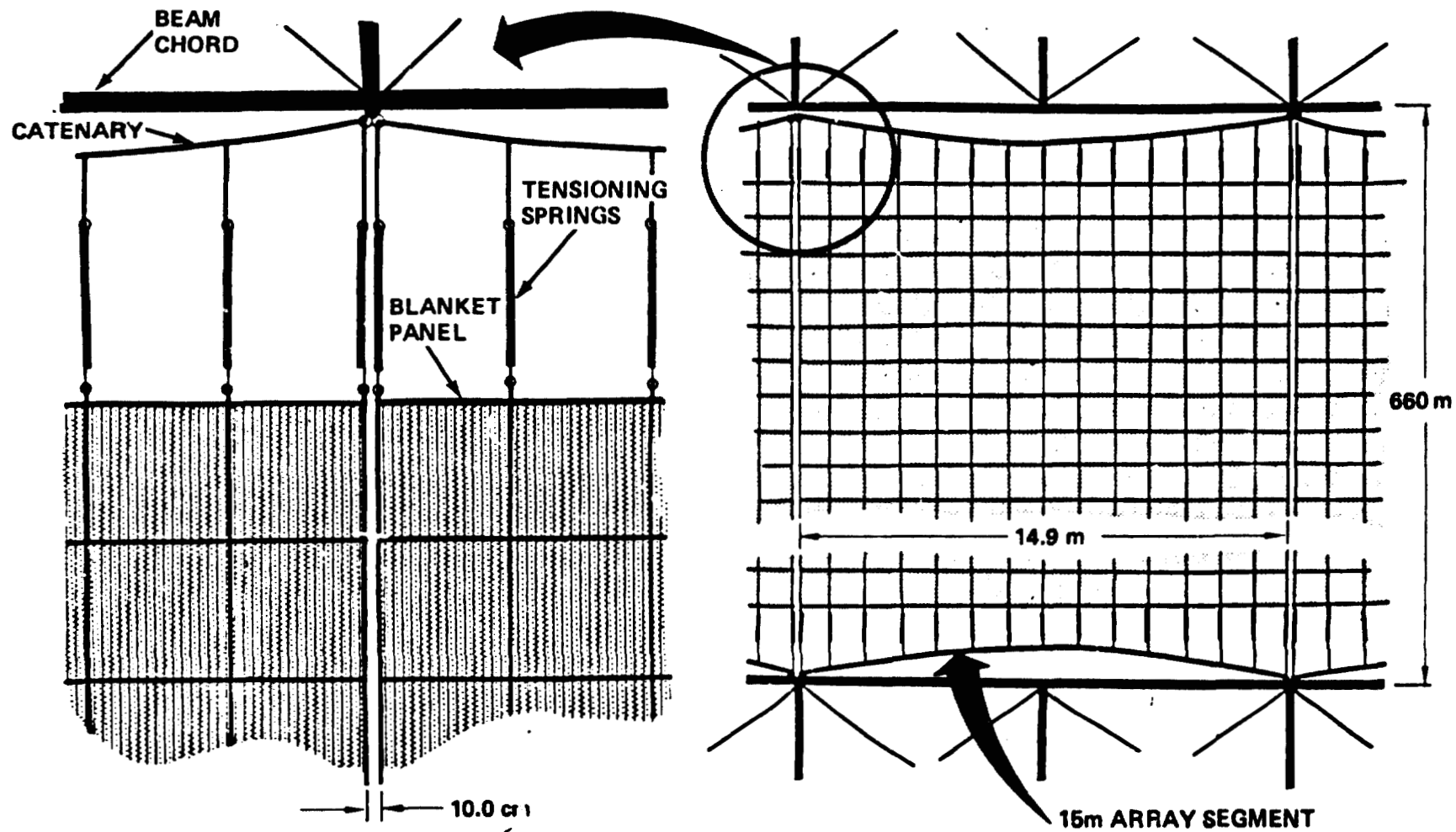
A uniaxial blanket support was selected over the biaxial support shown in Part II of this study. This change was the result of analysis of construction techniques and associated blanket uniformity problems. It will be necessary to provide batten tapes between blanket segments, at a few intervals along the segment length, to provide correct segment-segment orientation.

# Reference Array Blanket Support



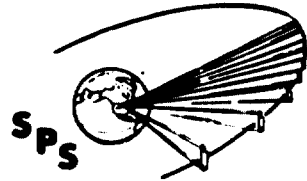
SPS-1843

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### **CONTINUOUS CHORD/BATTEN CONFIGURATION**

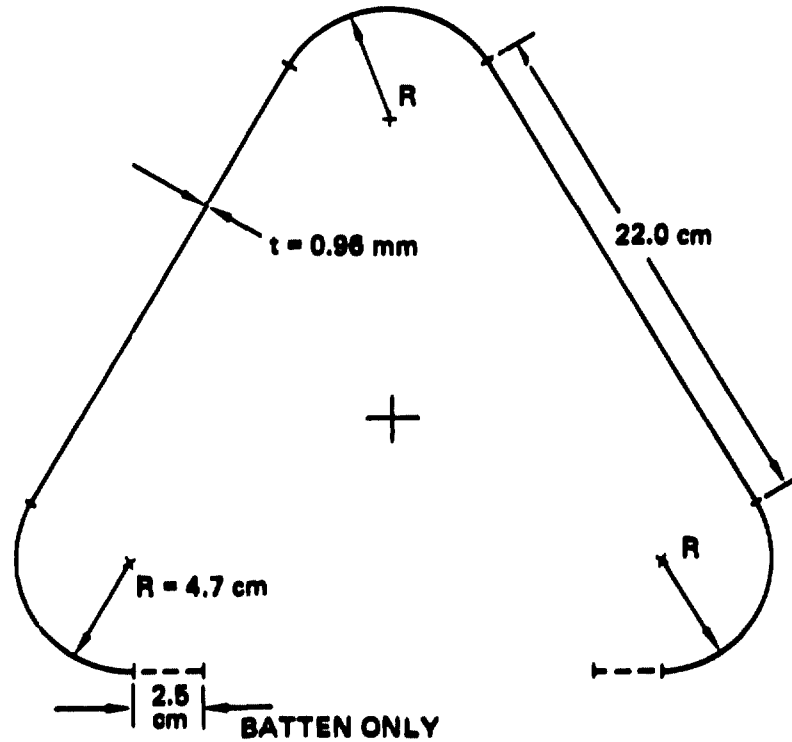
Shown here is the basic configuration and dimensions of the continuous chord beam elements. The load carrying capability of the continuous chord beam is very sensitive to  $EI_x$  of both the chord and batten. The analysis suggested that the chord and batten configuration should be the same for optimum performance. This is also beneficial from the fabrication standpoint since the same type of equipment can be used to make the chord and batten. The major differences between the chords and battens are that the battens have an additional 2.5 cm of material on each side of the open face for bonding to the chord and the battens are terminated at beam widths.



SPS-1992

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# Continuous Chord/Batten Configuration



**MATERIAL:** P-17C0 GRAPHITE (POLYSULFONE IMPREG)  
E-181 GLASS COVER

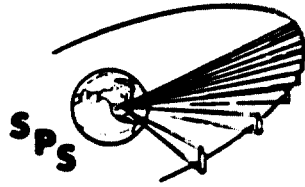
**BEAM:** WIDTH-7.5m  
BATTEN SPACING-7.5m  
MASS/LENGTH-5.54 kg/m

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### **CONTINUOUS CHORD BEAM APPROACH**

A comparison was made between the continuous chord beam approach and the tapered tube beam used in the Part II reference system. Solutions were found that met the load requirements of the tapered tube beam and resulted in a relatively small mass increase for the overall system.

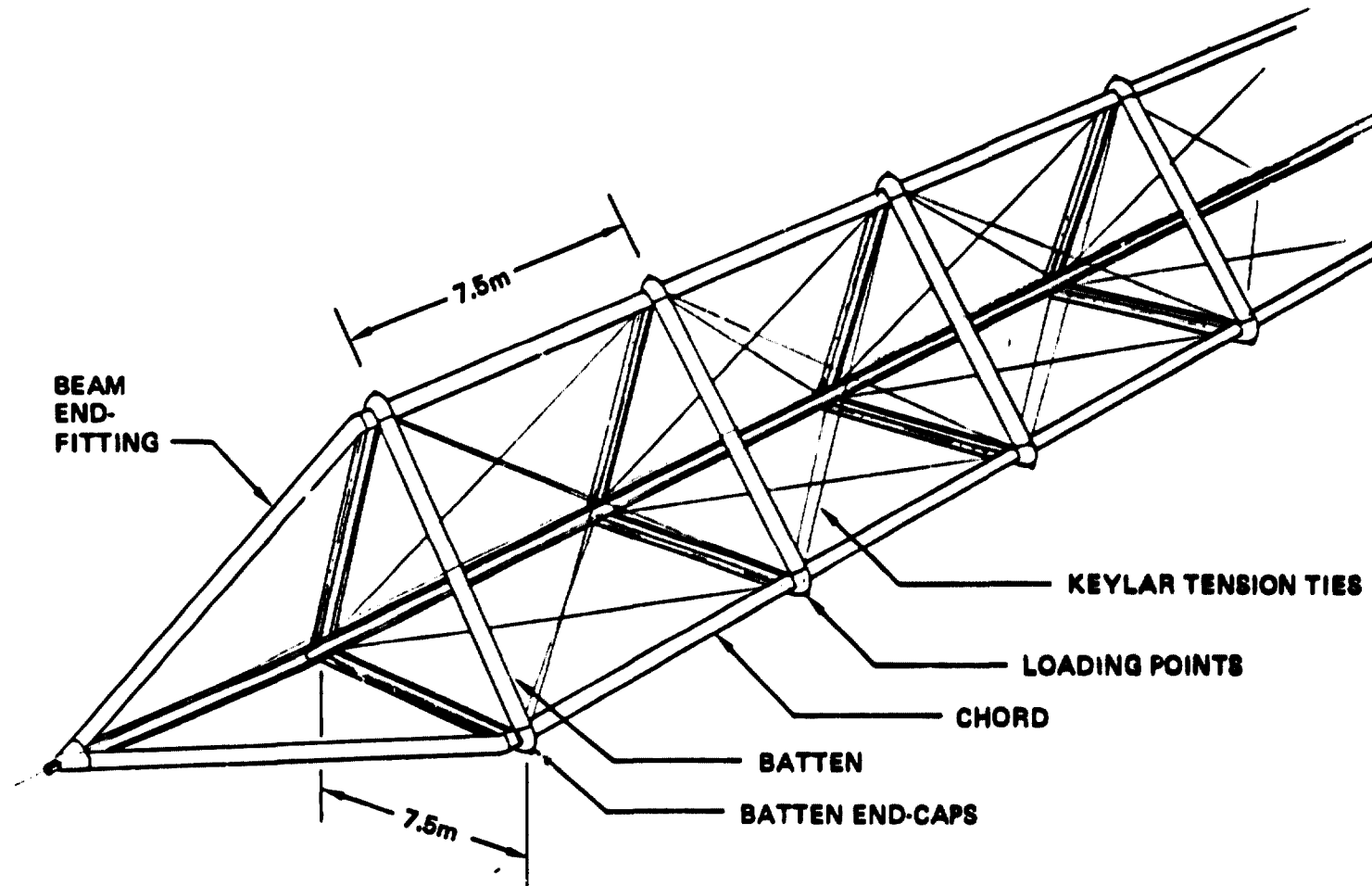
Shown here is a continuous chord beam approach that satisfies load/stiffness requirements for the reference photovoltaic system. This approach has loading points that are consistent with the current solar blanket geometry and also provides for centroidal beam-to-beam load transmittal.



SPS-1845

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## Continuous Chord Beam Approach

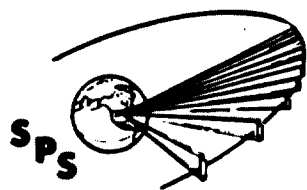


**D180-24071-3**

### **CONTINUOUS CHORD BEAM-BEAM INTERSECTION**

**Shown here is a typical satellite module edge joint. This type of joint permits centroidal beam-to-beam load transmittal.**

**This structural approach, with centroidal end-fittings, is consistent with current construction techniques and construction facility sizing.**

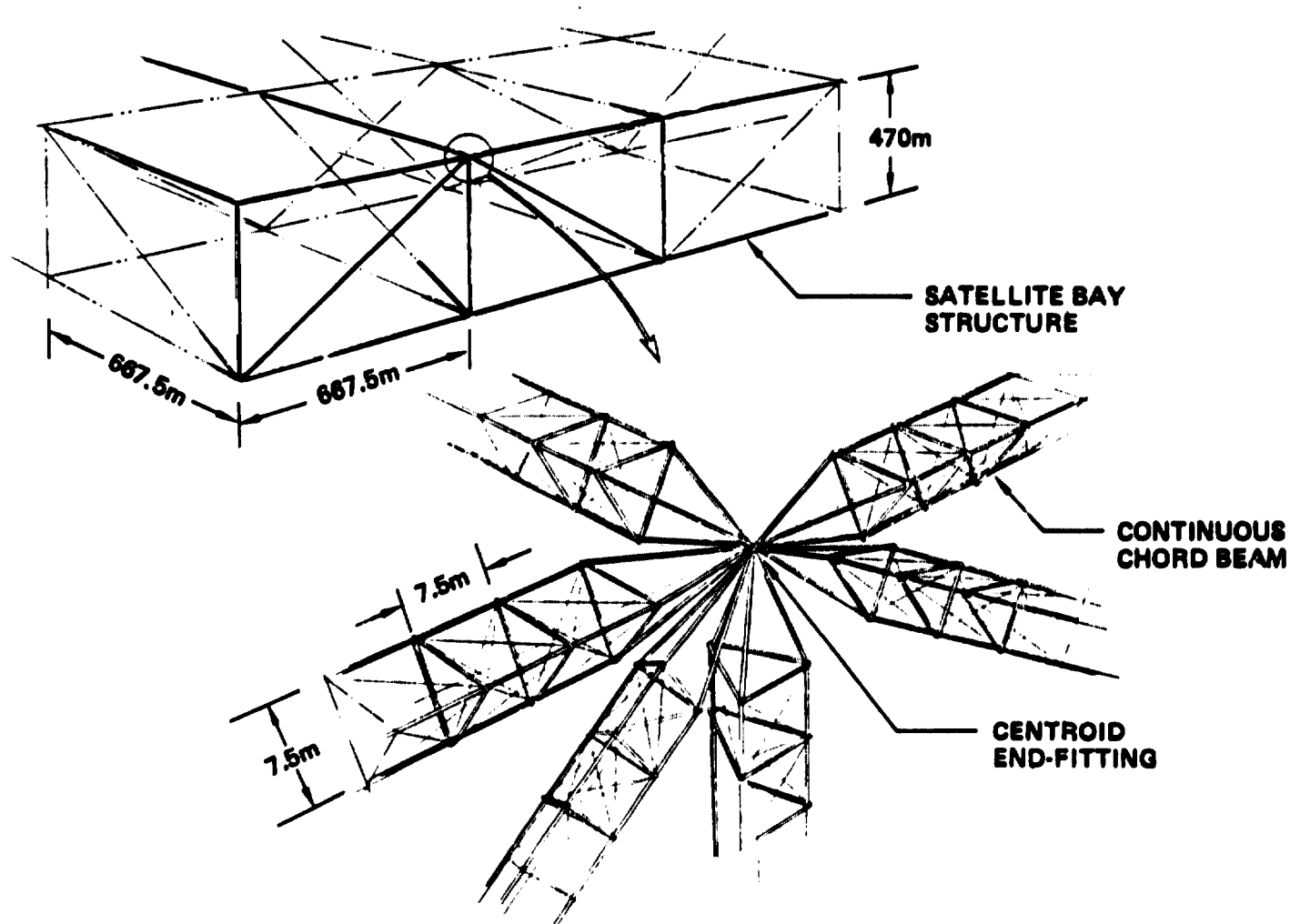


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## Continuous Chord Beam-Beam Intersections

SPS-1844

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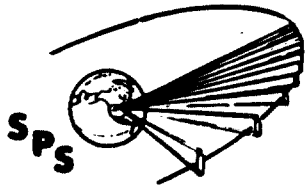


**REFERENCE MPTS STRUCTURAL CONCEPT**

Shown here is an illustration of the new orthogonal approach used for the MPTS reference system. This change was the result of construction/maintenance trades and involves the incorporation of a klystron module as the system LRU.

The mechanical and electrical rotary joints are the same as that shown at the Part II final review. However, the interface between the satellite primary structure and the mechanical rotary joint has been changed to provide for better load transmittal.

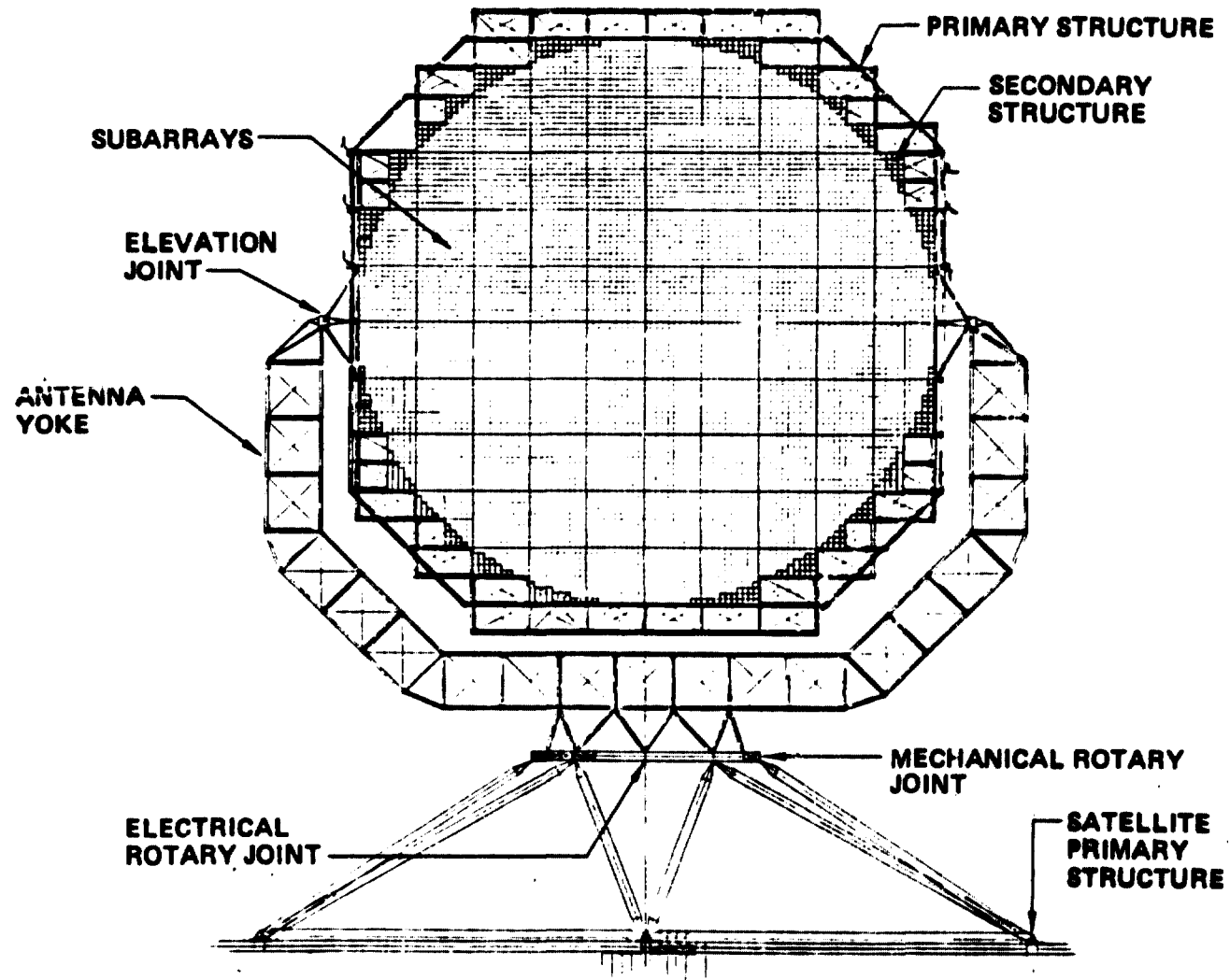
Other changes have occurred that are reflected in the antenna geometry and in the yoke on which it is mounted. These changes are reflected down to the subarray level where a square matrix (10.43 m on a side) was used to provide the three point subarray support.



SPS-1993

BOEING

## Reference MPTS Structural Approach

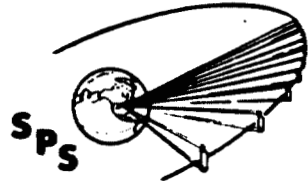


### **REFERENCE MPTS STRUCTURE INTERFACES**

The relative size and configuration of the MPTS system is shown in perspective. The primary structure gives the depth necessary for platform stiffness and provides support points for the cubic secondary structure. The secondary acts as an interface between the subarray and the primary structure. Each subarray is provided with three support points on the secondary structure to allow the necessary adjustments for array flatness and pointing ability.

Power converters will be located on the back of the primary structure. The electrical busing will run along the primary beams and be distributed at the secondary level to provide power to the subarrays/klystrons. Power converter thermal control equipment will also be located on the back of the primary structure.

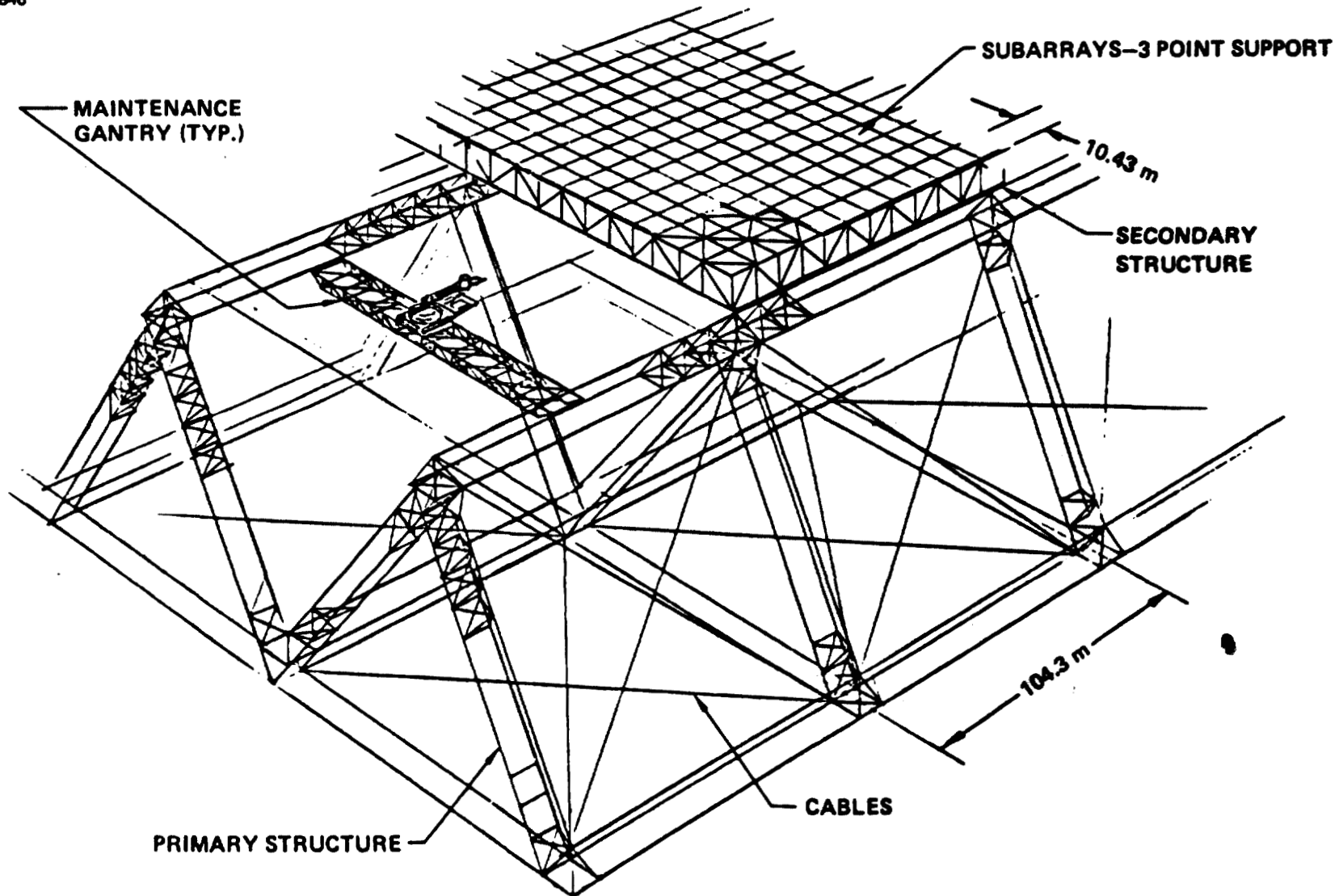
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SPS-1846

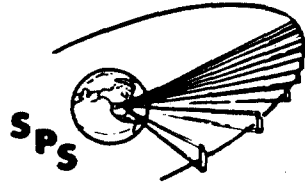
BOEING

## Reference MPTS Structure Interfaces



**POWER TAPER INTEGRATION**

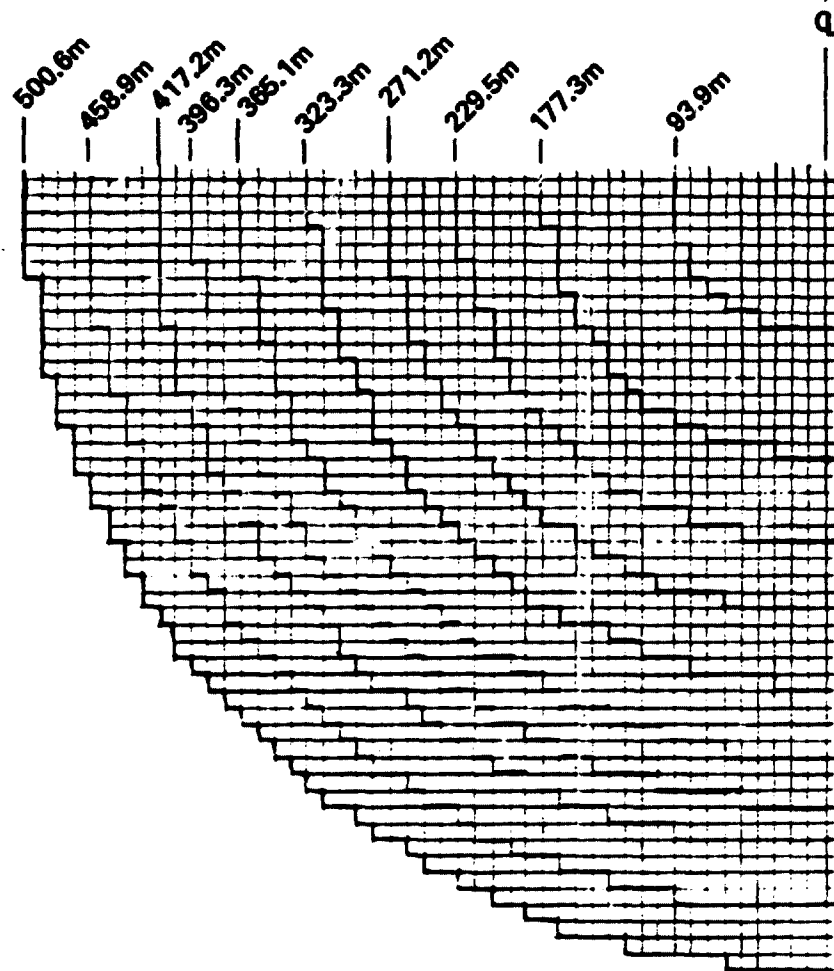
The new MPTS baseline uses a square subarray instead of the rectangular subarray shown in Part II. It was necessary to iterate this into the integration of the power taper on the MPTS array. The actual integration of power density rings is illustrated on this view of one-fourth of the radiating face of the antenna. The integration simulates a gaussian power taper of 9.5 dB using the quantized power levels available. Note the change in numbers of subarrays and klystrons over the Part II reference system.



SPS-1842

# MPTS Reference Power Taper Integration

BOEING



| STEP   | NUMBER<br>SUBARRAYS | NUMBER<br>KLYSTRONS/<br>SUBARRAYS | NUMBER<br>KLYSTRONS |
|--------|---------------------|-----------------------------------|---------------------|
| 1      | 276                 | 36                                | 9,936               |
| 2      | 632                 | 30                                | 18,960              |
| 3      | 644                 | 24                                | 15,456              |
| 4      | 628                 | 20                                | 12,560              |
| 5      | 784                 | 16                                | 12,544              |
| 6      | 900                 | 12                                | 10,800              |
| 7      | 664                 | 9                                 | 5,976               |
| 8      | 612                 | 8                                 | 4,896               |
| 9      | 1,052               | 6                                 | 6,312               |
| 10     | 1,028               | 4                                 | 4,112               |
| Totals | 7,220               |                                   | 101,552             |

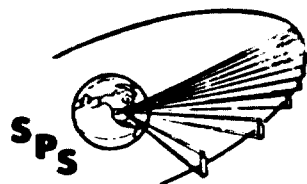
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### **INTEGRATED SUBARRAY**

The same basic layout of subarray components, used in the Part II final review, is used in the current reference. The only changes that have occurred are in the basic subarray geometry (square instead of rectangular) and in the method of klystron support within the module.

The square subarray was the result of going to an orthogonal support structure for an improved maintenance approach over the triangular support shown previously.

The new method of klystron support within the subarray reflects the new module LRU approach. To facilitate klystron module removal (excluding the radiating waveguide) it was necessary to shorten the klystron support C-beam and provide a support block for load transmittal into the waveguide. In this way the klystron, output waveguide, thermal control, control circuitry, and support mechanism can be removed as an integral unit. This provides for an improved maintenance scheme over that shown previously.

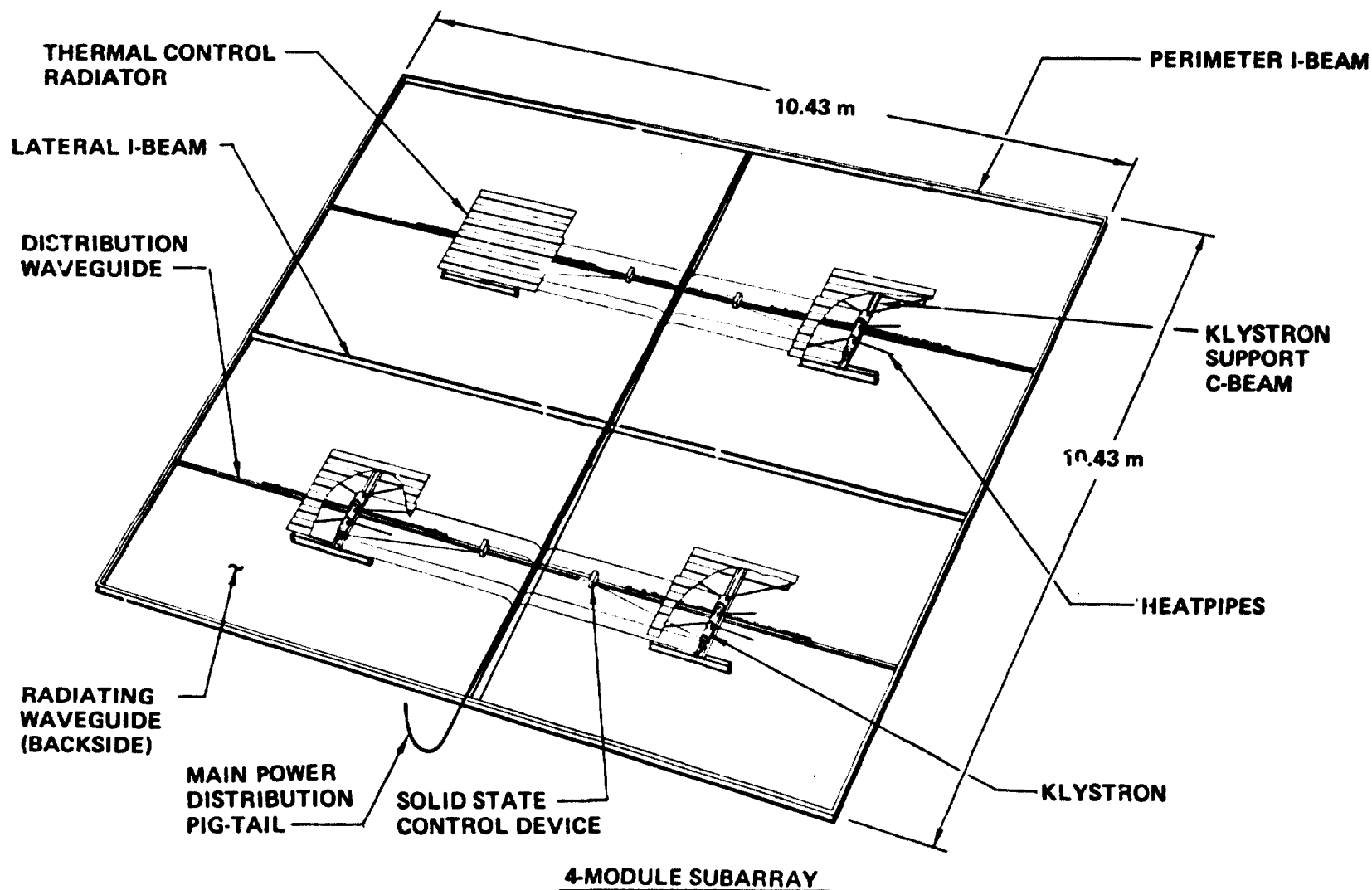


D180-24071-3

# Reference MPTS—Integrated Subarray

SPS-1217

BOEING

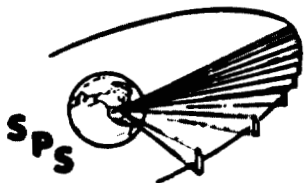




### **FEATURES OF BASELINE RETRODIRECTIVE PHASE CONTROL**

The phase control system utilizes a 3 node phase distribution network similar in some respects to the distribution network earlier suggested by Dickinson of JPL. To minimize phase error buildup, only three nodes are used. This network feeds phase control to a total of 9614 subarrays.

At the lower left is shown the frequency plan for the pilot beam system. The pilot beam is a double sideband, suppressed carrier, AM modulated system. The suppressed carrier is slightly offset from the power beam, and the pilot beam sidebands are either side of the power beam. This frequency plan avoids certain types of errors that would arise with only a single frequency pilot beam. At the lower right is shown the 3 pilot transmitter system located at the rectenna. The use of a 3 pilot transmitter allows the creation of a virtual phase center, at the center of the rectenna, that can be moved to correct for pointing errors. This adds another degree of flexibility to the phase control system that may be necessary to compensate for ionosphere disturbances.

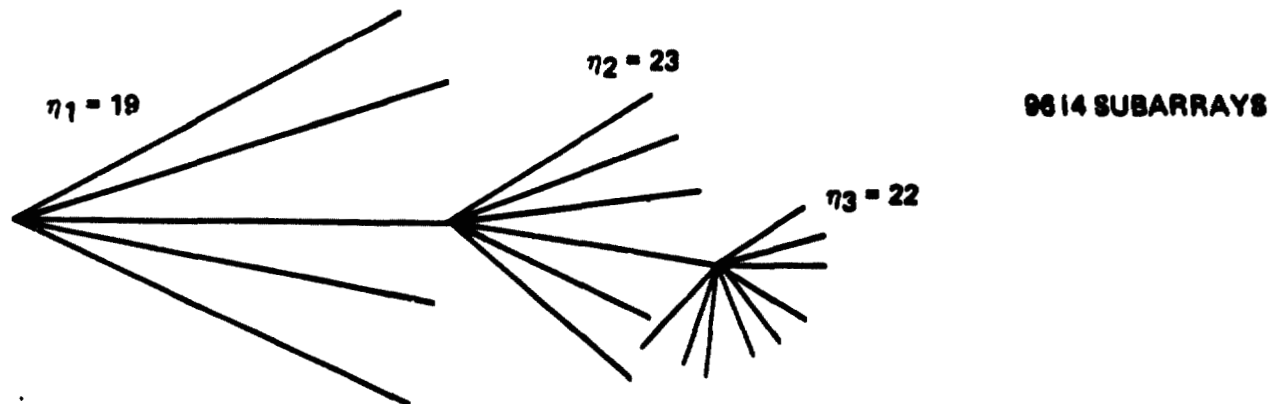


SPS-1865

D180-24071-3

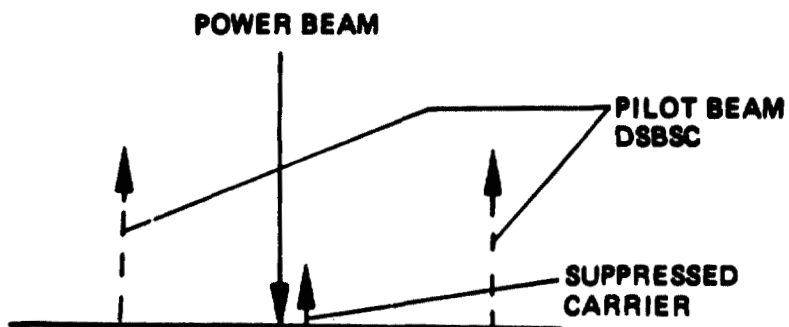
# Features of Baseline Retrodirective Phase Control

BEING

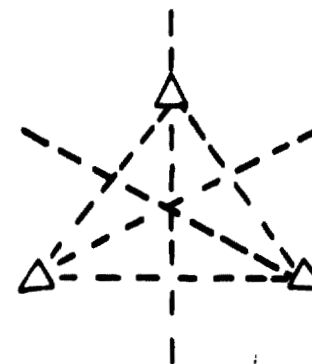


THREE NODE PHASE DISTRIBUTION SYSTEM FOR ERROR CONTROL

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DOUBLE SIDEBAND SUPPRESSED CARRIER  
AM MODULATED PILOT BEAM



3-PILOT ANTENNA TRANSMITTER

**PHOTOVOLTAIC REFERENCE CONFIGURATION NOMINAL MASS SUMMARY**

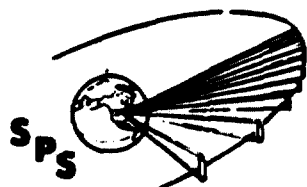
The structural mass difference from previous analysis reflects a change in the structural beam concept and integration of new sizing criteria to normalize the E.O.L. ground output to 10.0 GW. The majority of the structural mass increase is due to the lower load-to-mass capability of continuous chord elements. A small increase can also be attributed to the slight increase in satellite bay size to accommodate the increase in solar array necessary for normalizing power.

An increase in solar array area, to normalize the power to 10.0 GW, is reflected in the increase of solar cell blanket mass.

The small increase in power distribution mass can be attributed to the increased length of main buses caused by increased bay size.

MPTS mass increased to reflect the increased inventory of klystrons and power conversion equipment to normalize ground output power to 10.0 GW, using the current efficiency chain.

Approximately two thirds of the subtotal mass increase from Part II can be attributed to normalizing the power output to 10.0 GW. The remainder of the mass increase was caused by reflecting the continuous chord beam approach. It is interesting to note that the new structural approach did not significantly change the overall system mass.



D180-24071-3

# Photovoltaic Reference Configuration Nominal Mass Summary Weight in Metric Tons

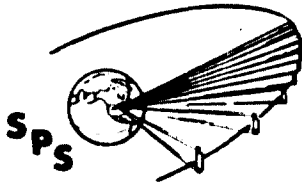
SPS-1906

BOEING

| COMPONENT                                 | PART II<br>FINAL | CURRENT         | REMARKS                                             |
|-------------------------------------------|------------------|-----------------|-----------------------------------------------------|
| <b>1.0 SOLAR ENERGY COLLECTION SYSTEM</b> | <b>(51,782)</b>  | <b>(55,602)</b> |                                                     |
| 1.1 PRIMARY STRUCTURE                     | 5,385            | 7,155           | CONTINUOUS CHORD BEAMS AND<br>NORMALIZING POWER     |
| 1.2 SECONDARY STRUCTURE                   | —                | —               |                                                     |
| 1.3 MECHANICAL SYSTEMS                    | 67               | 67              | NO CHANGE                                           |
| 1.4 MAINTENANCE STATION                   | —                | —               |                                                     |
| 1.5 CONTROL                               | 178              | 178             | NO CHANGE                                           |
| 1.6 INSTRUMENTATION/<br>COMMUNICATIONS    | 4                | 4               | NO CHANGE                                           |
| 1.7 SOLAR-CELL BLANKETS                   | 43,750           | 45,773          | INCREASED ARRAY AREA TO<br>NORMALIZE POWER TO 10 GW |
| 1.8 SOLAR CONCENTRATORS                   | —                | —               |                                                     |
| 1.9 POWER DISTRIBUTION                    | 2,398            | 2,425           | SLIGHT INCREASE IN TRANS-<br>MISSION LENGTH         |
| <b>2.0 MPTS</b>                           | <b>25,212</b>    | <b>26,379</b>   | <b>NORMALIZED POWER AND<br/>SQUARE SUBARRAY</b>     |
| <b>SUBTOTAL</b>                           | <b>76,994</b>    | <b>81,998</b>   |                                                     |
| <b>GROWTH</b>                             | <b>20,400</b>    | <b>17,590</b>   | <b>NORMALIZED POWER ON<br/>GROWTH CURVE</b>         |
| <b>TOTAL</b>                              | <b>97,474</b>    | <b>99,588</b>   |                                                     |

**MASS/SIZE UNCERTAINTY UPDATE**

Shown here is the current and Part II reference point designs compared to the three sigma mass/size uncertainty ellipse. The increase in size and mass of the current reference point design is attributed only to normalizing the system output to 10.0 GW.

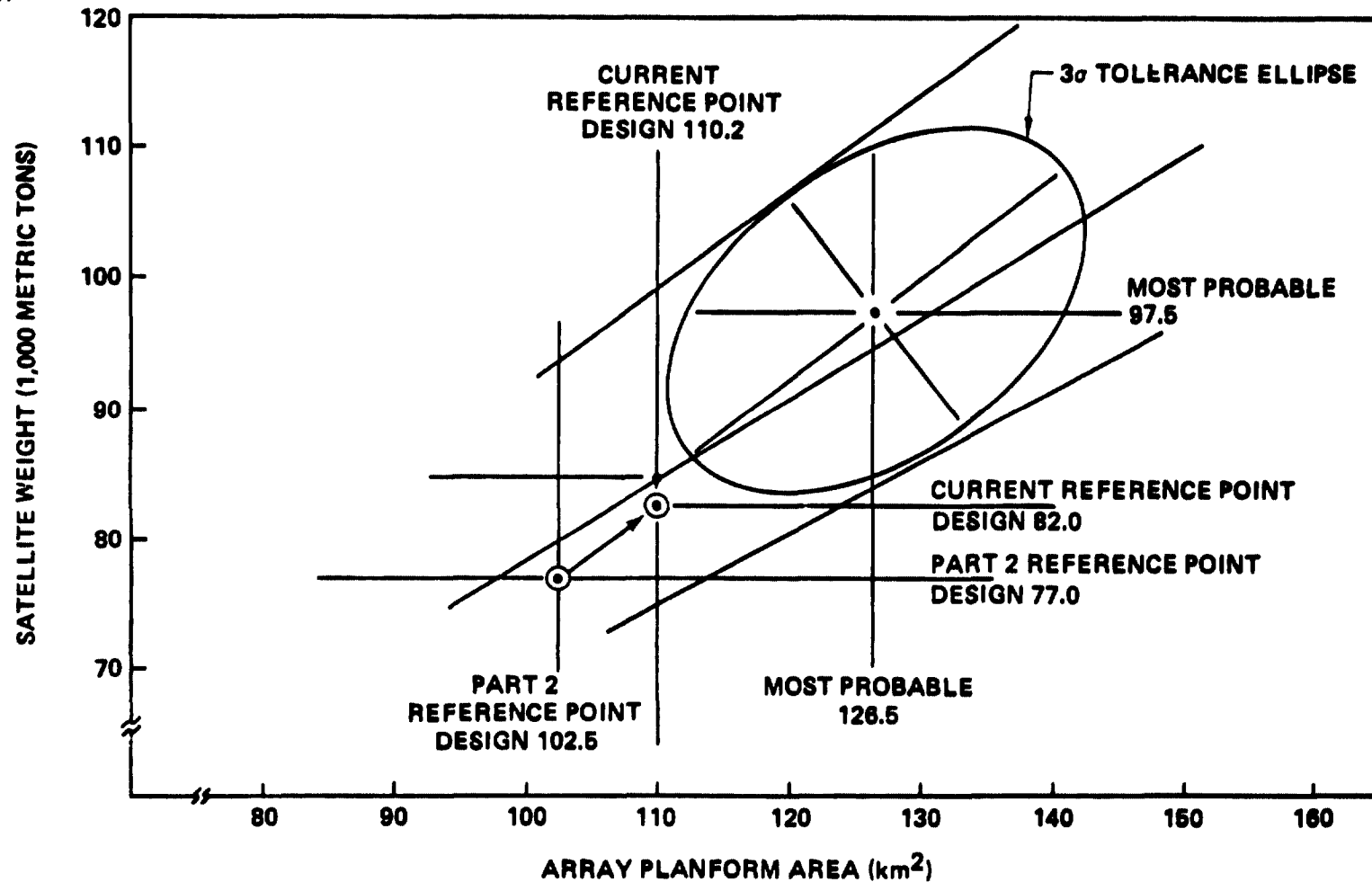


D180-24071-3

## Mass/Size Uncertainty Update

SPS-1994

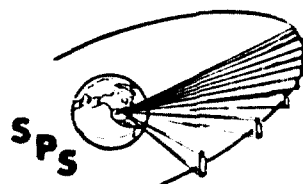
BOEING



**D180-24071-3**

### **COST UPDATE**

The Part II capital cost estimate was updated to reflect Part III changes. The principal new items were the addition of a grid interface system and an allotment of initial spares. Also, the power capability was renormalized to 10 gigawatts. Because the power renormalization was less expensive than predicted by the uncertainty analysis, the Part III capital cost/kwe changed little from Part II.



D180-24071-3

# Capital Cost Update Summary: 1 SPS Per/Year (In Millions of 1977 \$)

SPS-1981

BOEING

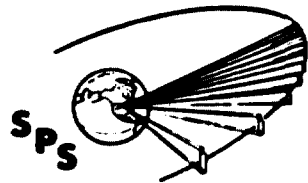
| ITEM                                    | PART II FINAL<br>(9.3 GW) | PART III UPDATE<br>(10 GW) | REASON FOR CHANGE                                                               |
|-----------------------------------------|---------------------------|----------------------------|---------------------------------------------------------------------------------|
| • SUPPORT SUBSYSTEMS                    | 697                       | 637                        |                                                                                 |
| • ENERGY CONVERSION<br>(SOLAR BLANKETS) | 3,750                     | 4,023                      | LARGER ARRAY FOR 10 GW                                                          |
| • POWER DISTRIBUTION                    | 133                       | 142                        | HIGHER POWER                                                                    |
| • MICROWAVE POWER<br>TRANSMISSION       | 2,622                     | 2,724                      | HIGHER POWER                                                                    |
| • GROUND RECEIVING<br>STATION (2)       | 4,442                     | 4,520                      | RE-ESTIMATE                                                                     |
| • GRID INTERFACE                        | —                         | 1,348                      | NOT INCLUDED IN PART II                                                         |
| • CONSTRUCTION & SPACE<br>SUPPORT       | 1,109                     | 1,109                      | —                                                                               |
| • SPACE TRANSPORTATION                  | 6,445                     | 6,387                      | INCREASED EARTH LAUNCH<br>COST BUT SAVINGS BY ORBIT<br>TRANSFER SYSTEM RECOVERY |
| • INITIAL SPARES                        | —                         | 240                        | NEGLECTED IN PART I                                                             |
| • PACKAGING & OTHER                     | 314                       | 602                        | INCREASED TO 5% OF<br>APPLICABLE ITEMS                                          |
| • INTEREST DURING<br>CONSTRUCTION       | 1,864                     | 2,082                      | HIGHER BASE COST                                                                |
| • GROWTH                                | 3,450                     | 3,115                      | SOME OF PART II GROWTH<br>INCLUDED POWER DEFICIENCY                             |
| TOTAL                                   | 24,766<br>(\$2,663/kWe)   | 26,929<br>(\$2,693/kWe)    |                                                                                 |

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**LEO CONSTRUCTION CONCEPT  
PHOTOVOLTAIC SATELLITE**

Eight modules and two antennas are constructed at the LEO base. All modules are transported to GEO using self-power electric propulsion. Two of the modules will transport an antenna while the remaining six modules will be transported alone. The GEO operation requires berthing (docking) the modules to form the satellite and deployment of the solar arrays not used for the transfer, followed by the rotation of the antenna into its desired operating position.



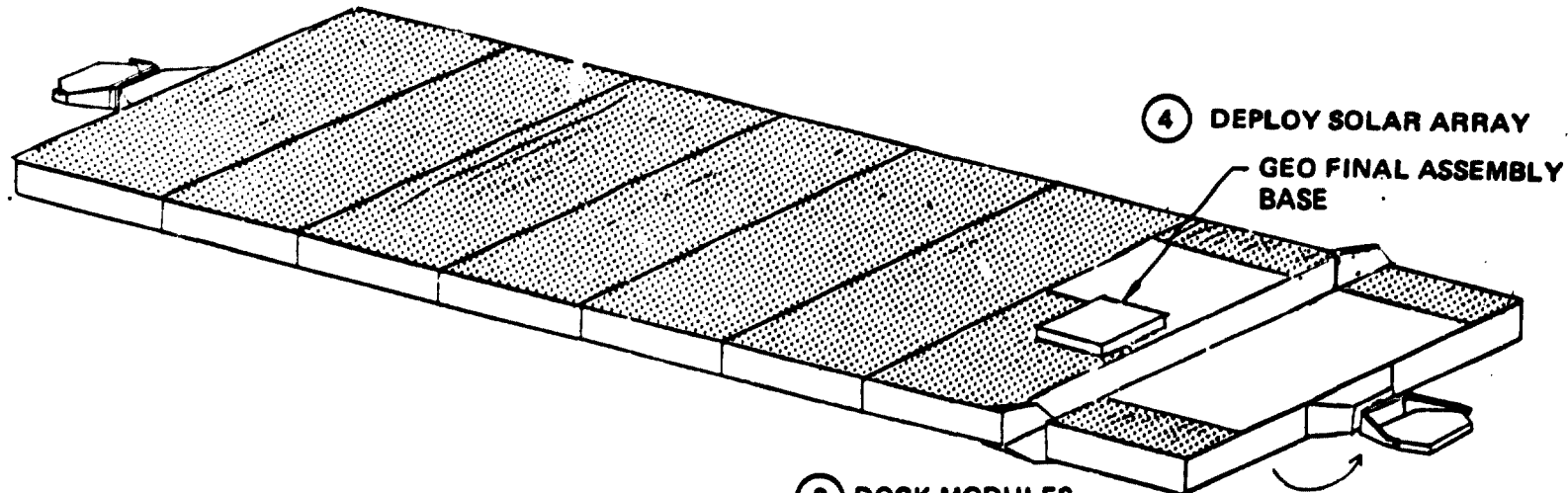
D180-24071-3

# LEO Construction Concept Photovoltaic Satellite

SPS-1383

BOEING

GEO



LEO  
CONSTRUCTION  
BASE

3 DOCK MODULES

5 ROTATE  
ANTENNA  
INTO  
POSITION  
(MODULES  
4 AND 8)

LEO

1 CONSTRUCT 8 MODULES  
AND 2 ANTENNAS

2 SELF POWERED  
TRANSPORT  
TO GEO (180 DAYS)

DEPLOY  
PORTION OF  
SOLAR  
ARRAY

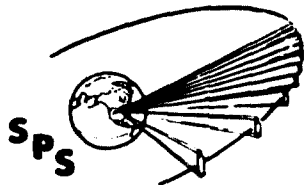
ANTENNA  
CARRIED  
BELOW  
(MODULES  
4 AND 8)

**LEO CONSTRUCTION BASE  
PHOTOVOLTAIC SATELLITE**

The construction base for the photovoltaic satellite consists of two connecting facilities with one used to build the modules and the other to build the antenna. The module construction facility is an open ended structure which allows the four bay wide module to be constructed with only longitudinal indexing. There are two internal working bays. The aft bay is used for structural assembly using beam machines and joint assembly machines attached to both the upper and lower surfaces of the facility. Solar array and power distribution are primarily installed from equipment attached to the lower facility surface in the forward bay. The satellite module is supported by movable towers located on the upper surface of the facility. These towers are also used to index the module as it is being fabricated.

The antenna facility is configured to enclose five bays of antenna in width and one row of bays in length. The antenna facility shown reflects the new antenna configuration. The upper surface of the facility is used to support beam machines, joint assembly machines, support indexing machines and bus deployment equipment. The lower surface is used to support beam machines, joint assembly machines and a deployment platform that is used to deploy the secondary structures and antenna subarrays.

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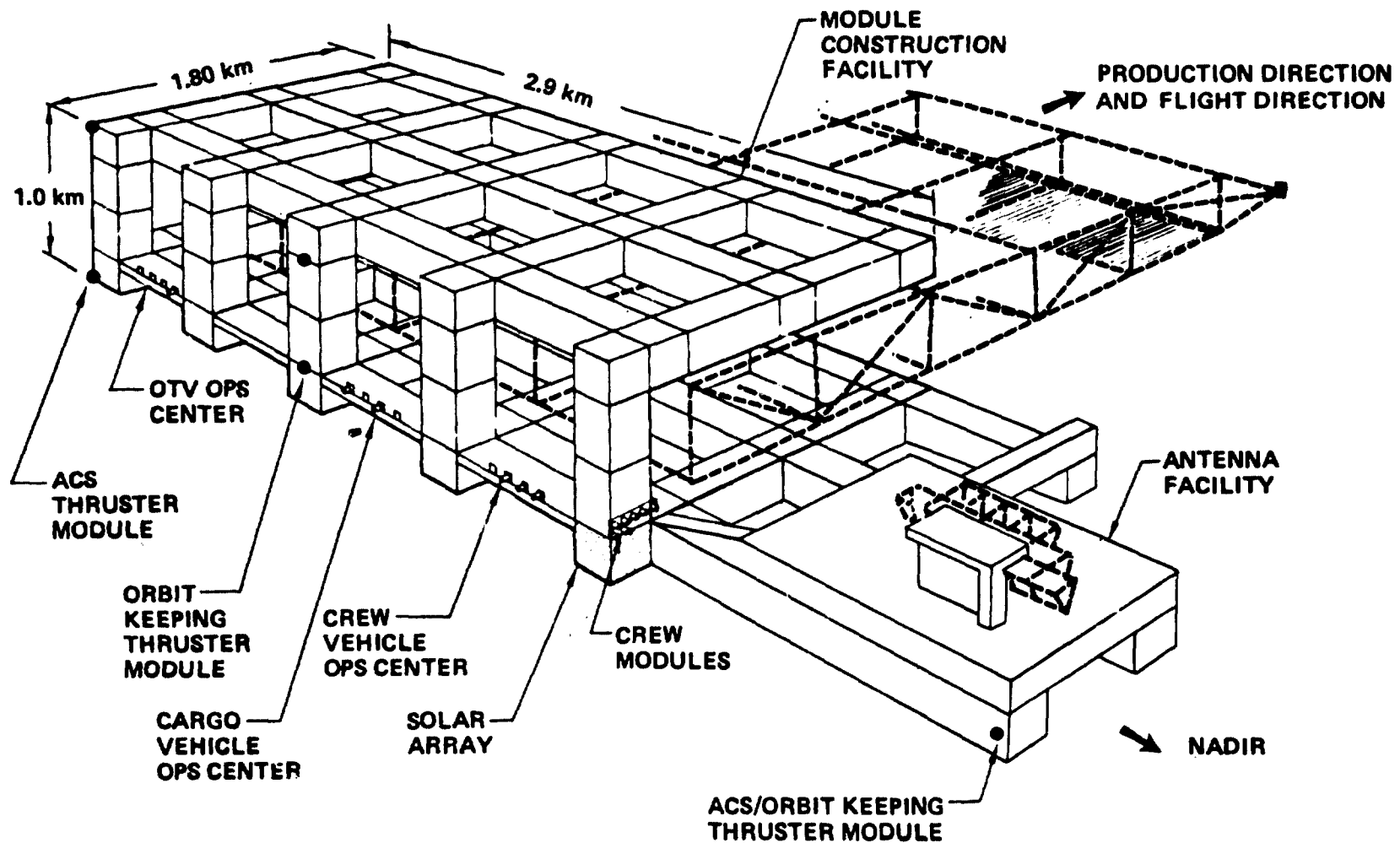


D180-24071-3

# LEO Construction Base Photovoltaic Satellite

SPS-1921

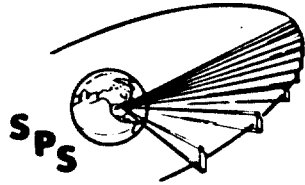
BOEING



### **CONSTRUCTION BASE EQUIPMENT AND OPERATION**

One of the principal changes in the construction base since Part II was to “invert” the satellite module with respect to the construction base to bring the solar blanket installation area closer to the payload arrival area. Most of the tonnage moved by the logistics network is solar blankets. It was felt important to minimize the distance over which these blankets are transported.

The other significant change was to modify the antenna construction facility to minimize the indexing and movement problems associated with installing the antenna in its yoke, also providing better crew access to the joint area where the antenna connects to the yoke.

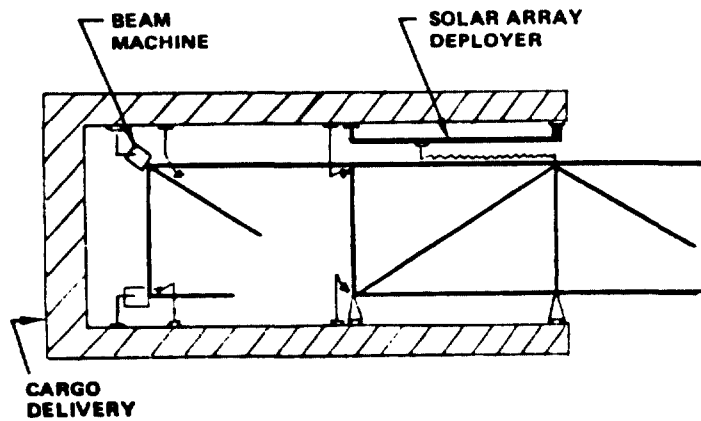


SPS-1926

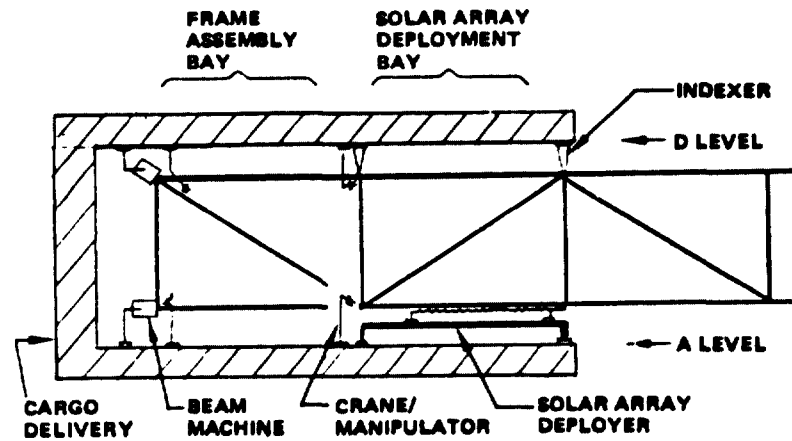
# Construction Base Equipment/Operations

BOEING

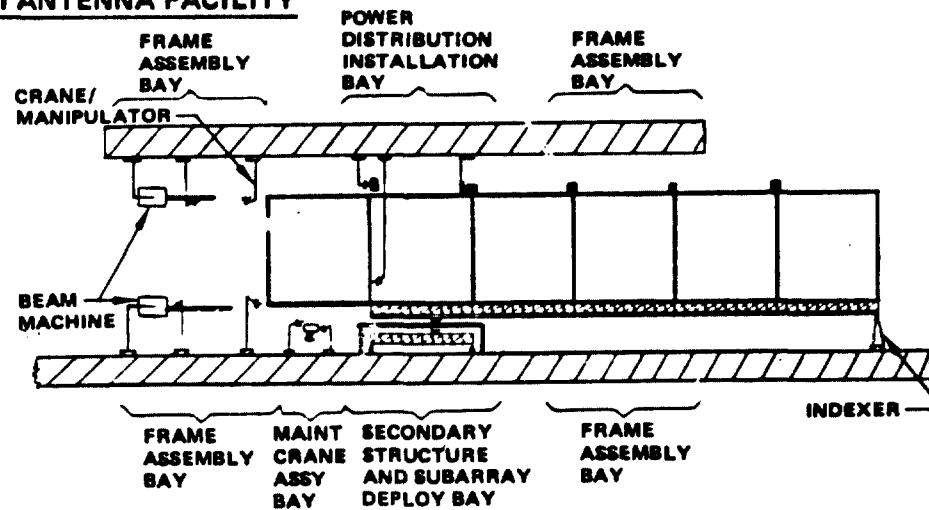
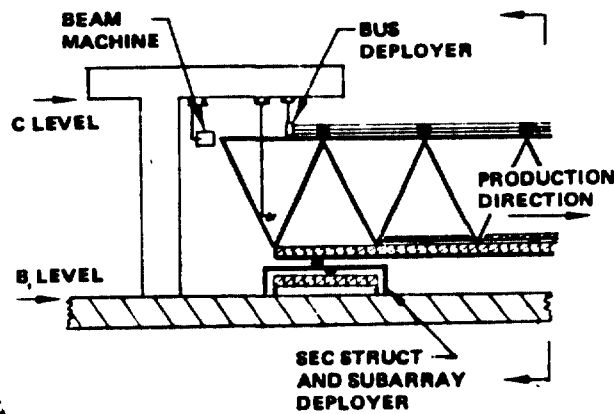
## PART II MODULE FACILITY



## PART III MODULE FACILITY



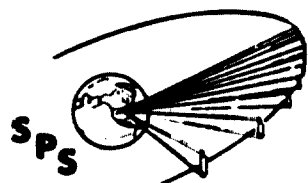
## PART III ANTENNA FACILITY



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**LOGISTICS NETWORK LEVEL AT LEO CONSTRUCTION BASE**

One of the significant requirements identified during Part II but not characterized was the need for a logistics network to move SPS hardware and construction equipment and crews around on the facility to bring them to the location where the work is actually being done. During Part III a logistic network was characterized. A part of the network is shown here. The tracks allow movement of construction equipment, crews, and satellite parts, and also provide for movement of the satellite module under construction by moving the indexing fixtures that tie the module to the facility.

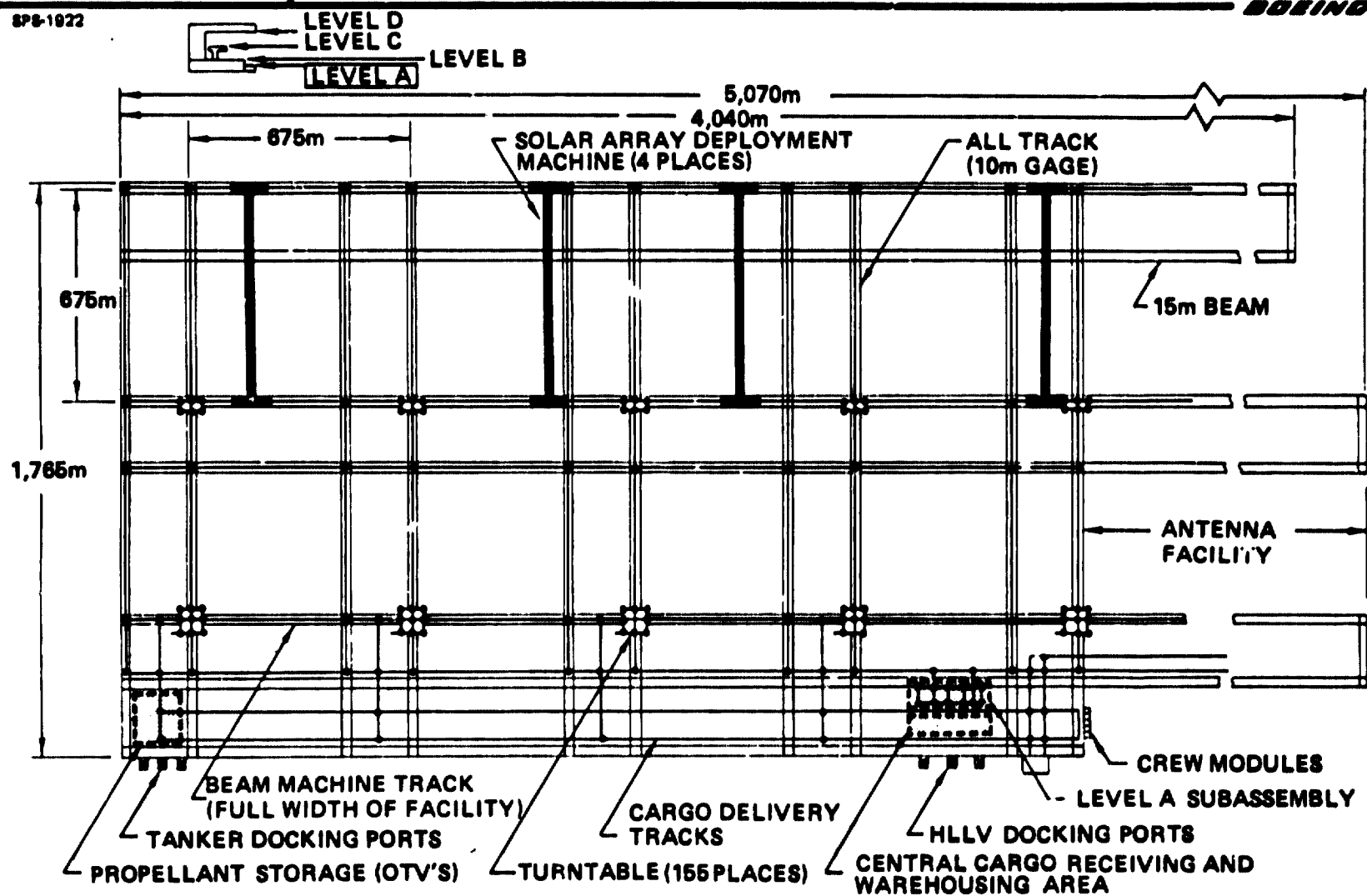


D180-24071-3

# Logistic Network Level A LEO Construction Base

SPS-1922

BOEING

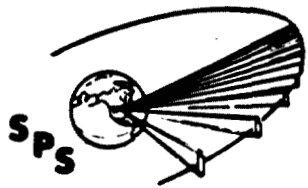


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### **GEO FINAL ASSEMBLY BASE AND OPERATIONS**

The maintenance analysis indicated a significant need for crews in geosynchronous orbit to carry out maintenance operations. The most straightforward way of providing for these crews appears to be to include their provision in the geosynchronous final assembly base and make this base also an operations base. The additions required include a klystron tube refurbishment facility, a docking location for the mobile maintenance habitat, and crew modules for the maintenance crews in addition to those for the final assembly base crews.

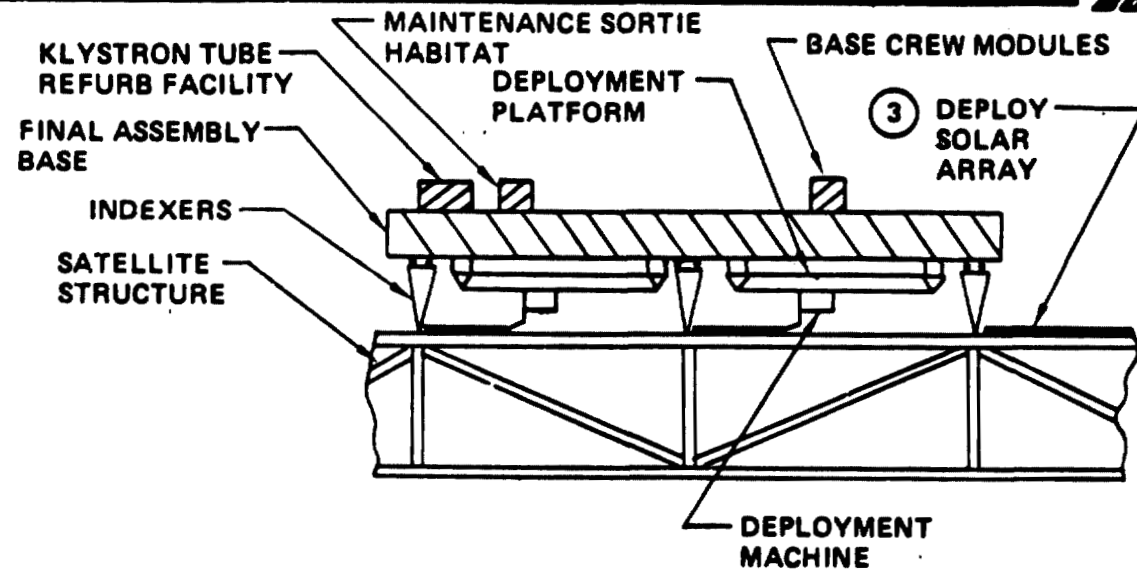


D180-24071-3

# GEO Final Assembly Base/Operations

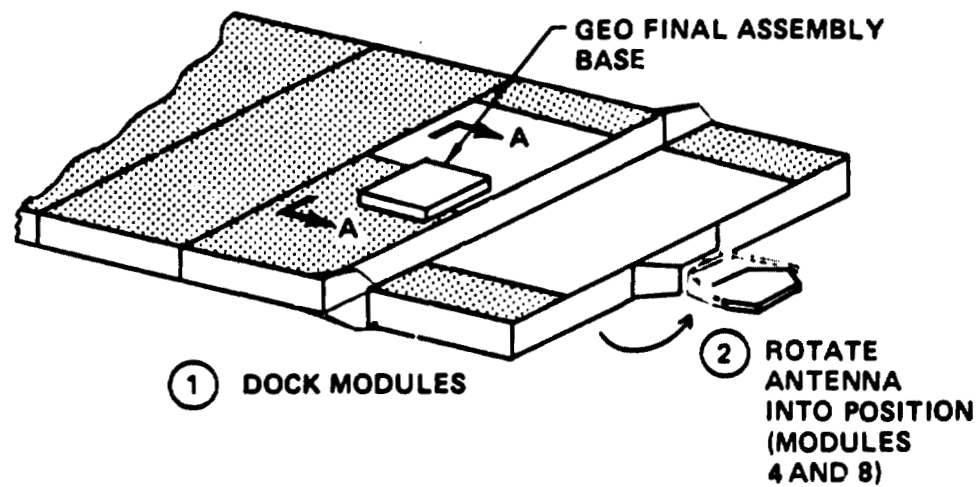
SPS-1412

BOEING



A-A

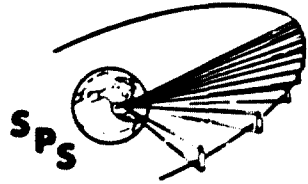
4 FINAL CHECKOUT



**D180-24071-3**

**CONSTRUCTION SYSTEMS CHARACTERISTICS**

Highlights of the construction base are given on this table.



SPS-1932

**BORING**

# Construction System Characteristics

|                          | LEO<br>CONSTRUCTION<br>BASE | GEO<br>FINAL ASSEMBLY<br>BASE |
|--------------------------|-----------------------------|-------------------------------|
| • CREW <sup>1</sup> ▶    | (480)                       | (65)                          |
| • CONSTRUCTION           | 200                         | 25                            |
| • CONSTRUCTION SUPPORT   | 140                         | 10                            |
| • OPERATIONS             | 140                         | 30                            |
| • MASS (MILLIONS OF Kg)  | (5.6)                       | (0.9)                         |
| • FACILITY               | 5.2                         | 0.7                           |
| • CONSTRUCTION EQUIPMENT | 0.4                         | 0.2                           |
| • DDT&E COST             | ( )                         | ( )                           |
| • FACILITY               |                             |                               |
| • CONSTRUCTION EQUIPMENT |                             |                               |
| • UNIT COST (BILLIONS)   | (4.8)                       | (1.2)                         |
| • FACILITY               | 3.5                         | 0.4                           |
| • CONSTRUCTION EQUIPMENT | 1.3                         | 0.4                           |
| • WRAP-AROUND            | 2.2                         | 0.4                           |

<sup>1</sup>▶ SATELLITE MAINTENANCE CREW NOT INCLUDED.

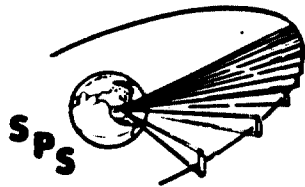
**TWO-STAGE WINGED SPS LAUNCH VEHICLE  
(FULLY REUSABLE CARGO CARRIER)**

The launch configuration of the SPS cargo vehicle is shown on the adjacent chart with the overall geometry noted. This series burn concept uses 16 LCH<sub>4</sub>/LO<sub>2</sub> engines on the booster and 14 standard SSME's on the orbiter. The LCH<sub>4</sub>/LO<sub>2</sub> booster engines are a gas generator cycle providing a vacuum thrust of  $9.79 \times 10^6$  newtons each. The SSME's on the orbiter provide a vacuum thrust of  $2.09 \times 10^6$  newtons (100% power level). The nominal 100% power level for the SSME's was selected based on engine life considerations which indicated about a 3 factor reduction in life if the 109% power level is used.

An airbreather propulsion system (12 installations of an SST type engine) has been provided on the booster for flyback capability to simplify the booster operational mode. The reference wing for both stages is

$$\begin{aligned} S_W (\text{Orbiter}) &= 1446 \text{ m}^2 \quad (15,560 \text{ ft}^2) \\ S_W (\text{Booster}) &= 2330 \text{ m}^2 \quad (25,080 \text{ ft}^2) \end{aligned}$$

Heat sink thermal protection system is provided on the booster and the Shuttle's Reusable Surface Insulation (RSI) is used on the orbiter.

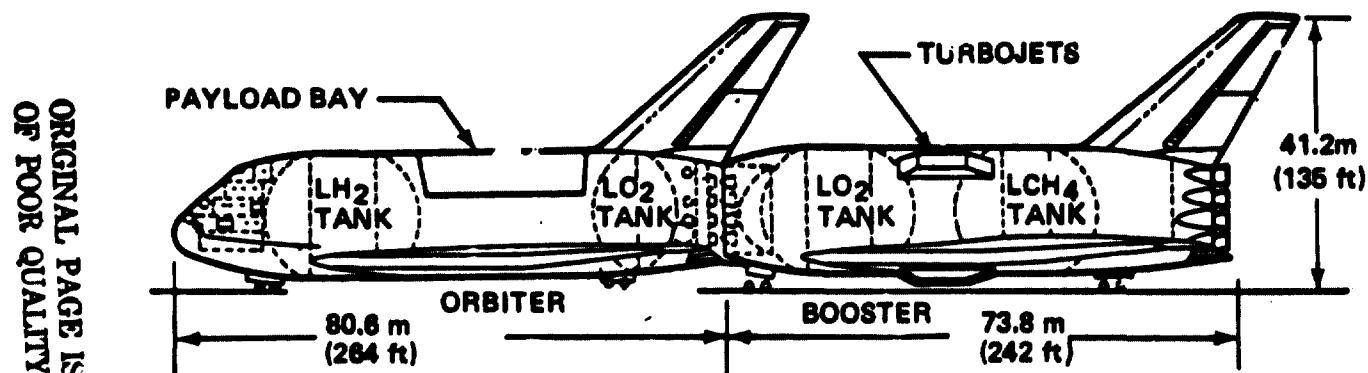
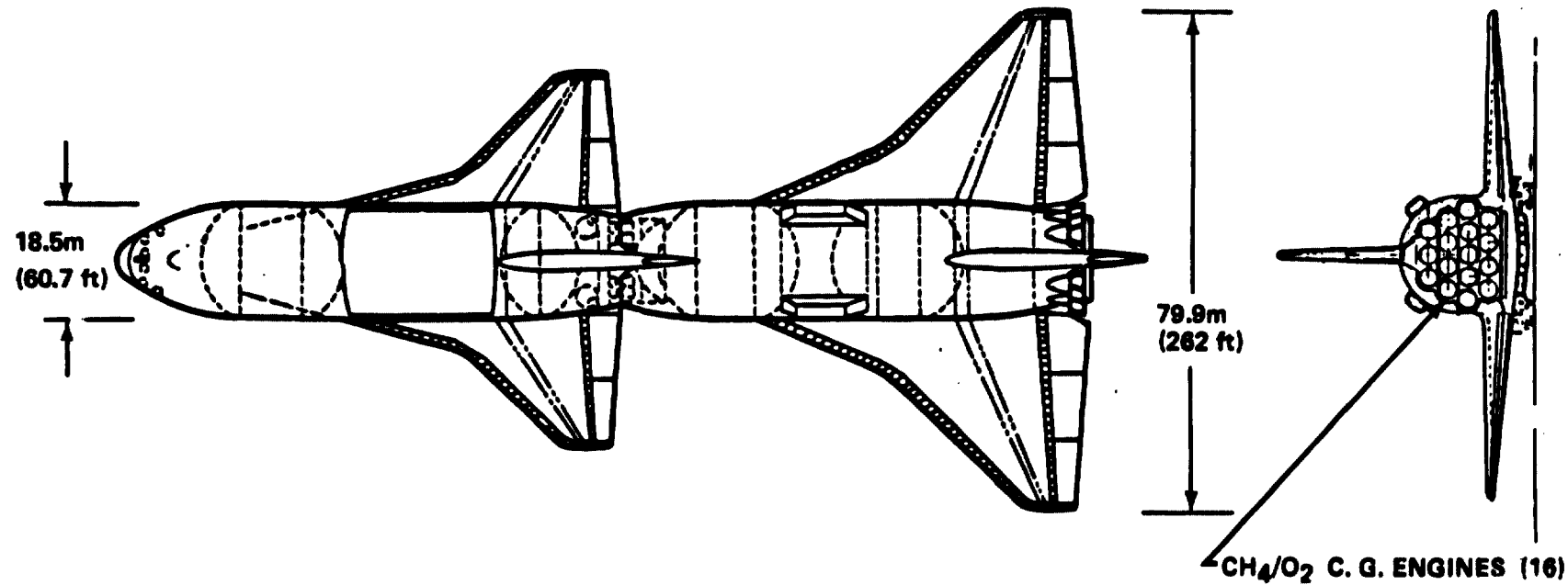


D180-24071-3

# Two-Stage Winged SPS Launch Vehicle (Fully Reusable Cargo Carrier)

SPS-1963

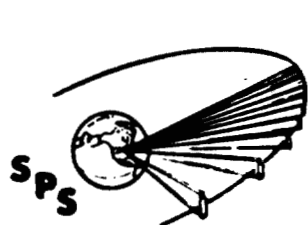
BOEING



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**CREW ROTATION AND RESUPPLY SYSTEM LEO/GEO APPLICATION**

Shown here is the oxygen/hydrogen orbit transfer vehicle and the payload module. This system provides the capability to transfer 75 crew, up and return, per flight with their supplies. The orbit transfer vehicle is a two stage, oxygen/hydrogen fueled, conventional rocket vehicle. The total propellant loading for the two stages is 460,000 kilograms.



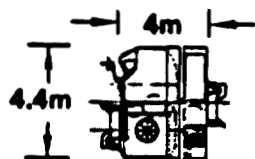
D180-24071-3

# Crew Rotation/Resupply System LEO/GEO Application

BOEING

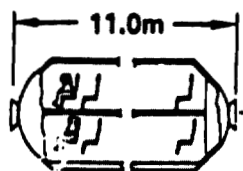
SPS-1939

## ● FLIGHT CONTROL MODULE



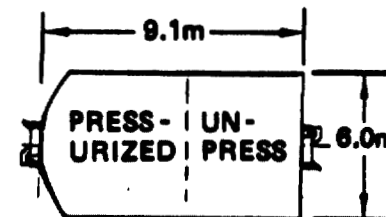
CREW = 2  
MASS = 4,000 kg

## ● GEO PASSENGER MODULE



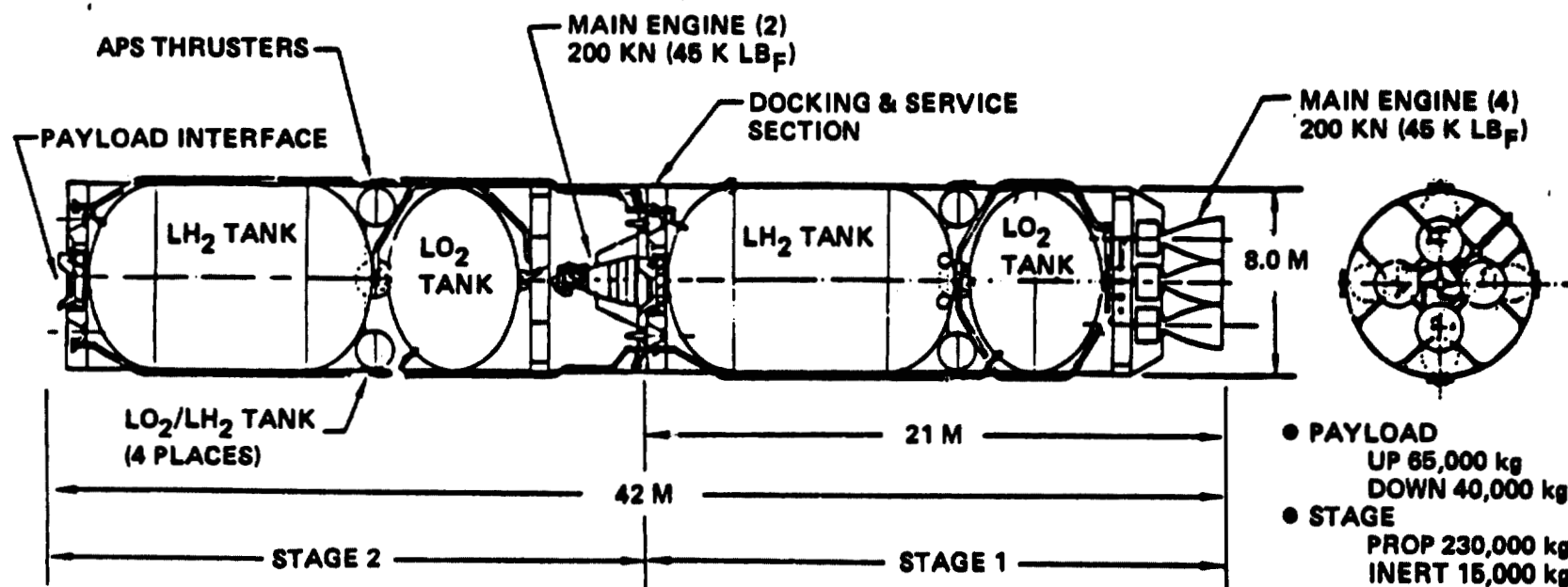
CREW = 75  
MASS = 20,000 kg

## ● SUPPLY MODULE



STAGE 2  
INTERFACE

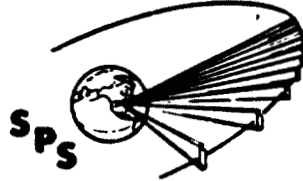
CARGO = 300 MAN MO.  
30,000 kg  
MODULE = 10,000 kg





**SELF-POWER CONFIGURATION  
PHOTOVOLTAIC SATELLITE**

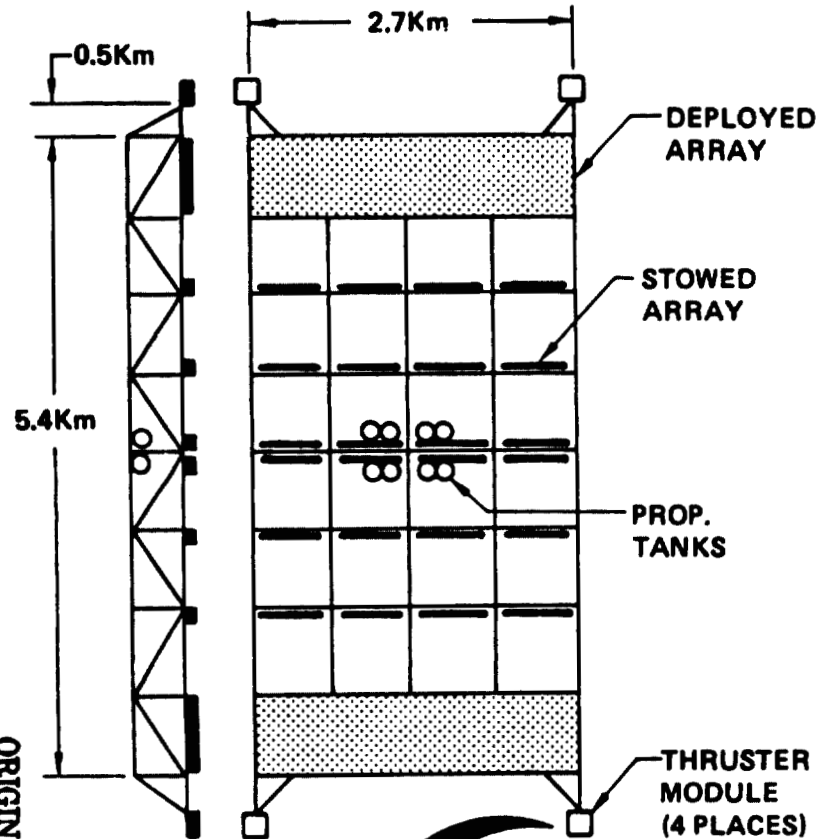
The transfer of the satellite modules from LEO to GEO involves the use of electric propulsion using power provided by the module (thus the name self-power). The characteristics associated with self-power of a photovoltaic module are shown for both those modules transferring antennas and those that do not. The general characteristics include a 5% oversizing of the satellite to compensate for the radiation degradation occurring during passage through the Van Allen belt and the inability to anneal out all of the damage after reaching GEO. It should also be emphasized at this point, only the arrays needed to provide the required power for transfer are deployed. The remainder of arrays are stowed within radiation proof containers. Cost optimum trip times and  $I_{sp}$  values are respectively 180 days and 7,000 seconds. Flight control of the module when flying a PEP attitude during transfer results in large gravity gradient torques at several positions in each revolution. Rather than provide the entire control capability with electric thrusters which are quite expensive, the electric system is sized only for the optimum transfer time with the additional thrust provided by  $LO_2/LH_2$  thrusters. This penalty actually is quite small since by the time 2,500 kilometer altitude is reached the gravity gradient torque is no longer a dominating factor.



# Self Power Configuration Photovoltaic Satellite

BOEING

SPS-1619



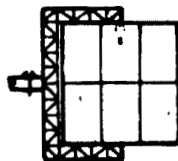
## GENERAL CHARACTERISTICS

- 5% OVERSIZING (RADIATION)
- TRIP TIME = 180 DAYS
- ISP = 7000 SEC

## MODULE CHARACTERISTICS

|                            | NO ANTENNA | WITH ANTENNA |
|----------------------------|------------|--------------|
| • NO. MODULES              | 6          | 2            |
| • MODULE MASS ( $10^6$ KG) | 8.7        | 23.7         |
| • POWER REQ'D ( $10^6$ Kw) | 0.3        | 0.81         |
| • ARRAY %                  | 13         | 36           |
| • OTS DRY ( $10^6$ KG)     | 1.1        | 2.9          |
| • ARGON ( $10^6$ KG)       | 2.0        | 5.6          |
| • $LO_2/LH_2$ ( $10^6$ KG) | 1.0        | 2.8          |
| • ELEC THRUST ( $10^3$ N)  | 4.5        | 12.2         |
| • CHEM THRUST ( $10^3$ N)  | 12.0       | 5.0          |

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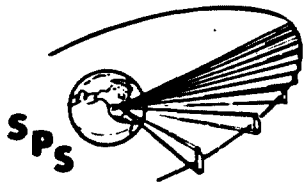


|                | NO ANTENNA | WITH ANTENNA |
|----------------|------------|--------------|
| PANEL SIZE:    | 24x38m     | 48x57m       |
| NO. THRUSTERS: | 560        | 1680         |

### RECOVERY CONCEPT FOR ELECTRIC OTS

The system analyzed for the return of the electric components to LEO is a single-stage  $\text{LO}_2/\text{LH}_2$  OTV. Return of 210,000 kilograms of payload requires a propellant loading of approximately 530,000 kilograms. This size of stage is slightly larger than that used for the crew rotation in the LEO construction option. When combined with a second stage, the combination vehicle serves to provide propulsion capability for resupply flights to GEO. Delivery of the  $\text{LO}_2/\text{LH}_2$  stage to GEO involves mounting the stages below the satellite module. The resulting impact on the electric propulsion system of transporting an additional 1.6 million kilograms of  $\text{LO}_2/\text{LH}_2$  stages is relatively minor.

Reuseability of the electric components used on the first module is not possible before transfer of the seventh module due to delivery times of 180 to 200 days.



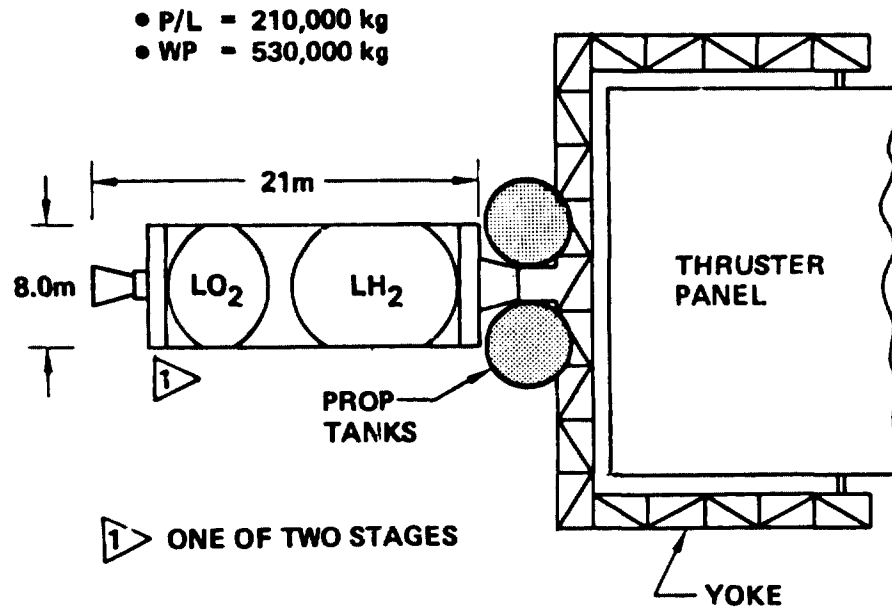
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# Electric OTS Recovery

SPS-1225

BOEING

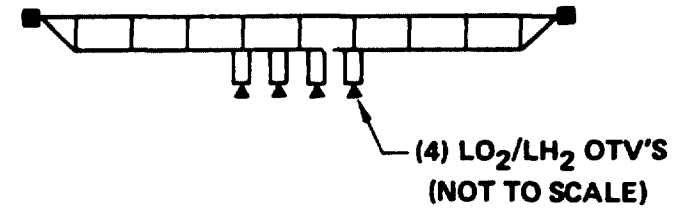
## RECOVERY SYSTEM



## COST SUMMARY

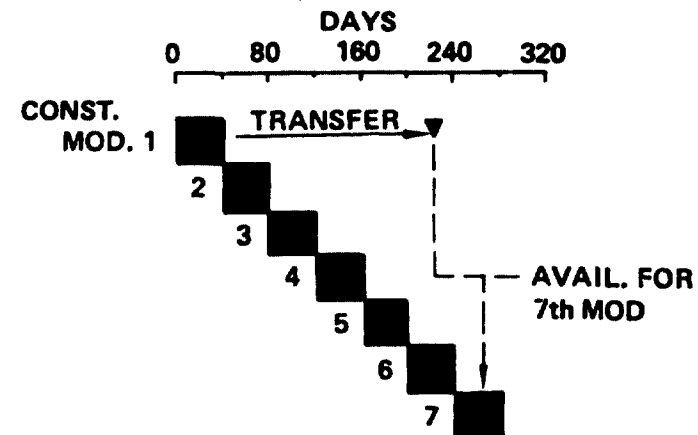
|                 | 1 SAT/YR | 4 SAT/YR |
|-----------------|----------|----------|
| • RECOV VALUE   | \$1,280M | \$690M   |
| • RECOV COST    | \$ 720M  | \$535M   |
| SAVINGS PER SAT | \$ 560M  | \$165M   |

## DELIVERY SYSTEM



- PIGGYBACK ON SATELLITE
- $\Delta$  P/L = 1.6 MILLION KG
- ELEC OTS MODIF  
 $\Delta$  POWER ~ 2%  
 $\Delta$  Wp ~ 0.4m Kg

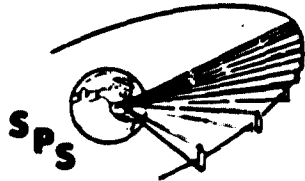
## OPERATIONS



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**A "CONSERVATIVE PACE" SPS SCENARIO**

This scenario represents a low-risk approach to SPS implementation. Major funding for the full size SPS system is not committed until successful on-orbit operation of a precursor satellite has been accomplished. This major funding includes not only that for the first full SPS but also that for development and implementation of the heavy lift launch vehicle fleet. Derivation of this scenario included allowances for the time period inherent in new vehicle development (five years for new stages, six years for new engines) and the time necessary for certain in-space operations. For example, the precursor program requires four and one half years from the first launch of equipment from which its construction base is assembled until the end of the test operations in geosynchronous orbit. SPS construction bases require two years of assembly and checkout operations prior to beginning SPS assembly. The self-power transfer of SPS modules to geosynchronous orbit takes 180 days.



# A "Conservative Pace" SPS Scenario

SPS 1983

BOEING

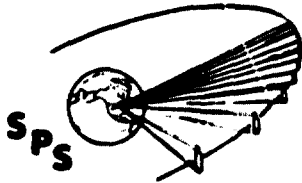
|                                               |    |    |    |    |                                       |    |    |    |    |                               |                |    |    |    |                            |                      |    |    |    |    |    |   |
|-----------------------------------------------|----|----|----|----|---------------------------------------|----|----|----|----|-------------------------------|----------------|----|----|----|----------------------------|----------------------|----|----|----|----|----|---|
| 78                                            | 79 | 80 | 81 | 32 | 83                                    | 84 | 85 | 86 | 87 | 88                            | 89             | 90 | 91 | 92 | 93                         | 94                   | 95 | 96 | 97 | 98 | 99 | 0 |
| <div>SPS TECHNOLOGY<br/>VERIFICATION</div>    |    |    |    |    |                                       |    |    |    |    |                               |                |    |    |    |                            |                      |    |    |    |    |    |   |
| LAUNCH<br>VEHICLES                            |    |    |    |    | ▼<br>SHUTTLE<br>DERIVATIVE<br>ON-LINE |    |    |    |    | ▼<br>HEAVY<br>LIFT<br>ON-LINE |                |    |    |    | ▼<br>HLLV<br>414 FLTS/YEAR |                      |    |    |    |    |    |   |
| CONSTRUCTION<br>BASES                         |    |    |    |    | ▼<br>PRECURSOR<br>BASE ON-LINE        |    |    |    |    | ▼<br>BASE 1<br>ON-LINE        |                |    |    |    | ▼<br>BASE 2<br>ON-LINE     |                      |    |    |    |    |    |   |
| SPS ON-LINE<br>(10 GW EACH, EXCEPT PRECURSOR) |    |    |    |    |                                       |    |    |    |    |                               | ▼<br>PRECURSOR |    |    |    |                            | 1/2 1 2 3<br>▼ ▼ ▼ ▼ |    |    |    |    |    |   |
| 78                                            | 79 | 80 | 81 | 82 | 83                                    | 84 | 85 | 86 | 87 | 88                            | 89             | 90 | 91 | 92 | 93                         | 94                   | 95 | 96 | 97 | 98 | 99 | 0 |

- NOTE: 1) THE PRECURSOR SPS SERVES AS A "COMMITMENT PRODUCER"
- 2) BY 2000, THE INSTALLED UNITS ARE PRODUCING ABOUT 11% OF THE ELECTRICAL ENERGY CONSUMED IN THE U.S. IN 1976.
- OF THE ELECTRICAL ENERGY CONSUMED IN THE U.S. IN 1976.

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**AN "APOLLO PACE" SPS SCENARIO**

The program represents a pace intermediate between that of the "conservative" scenario given on the previous chart and a "crash, Manhattan project". In this scenario development of the shuttle-derivative launch vehicle and the precursor **SPS** construction base are begun in 1979. Development of the heavy lift launch vehicle and other elements involved with the full size **SPS** starts before completion of the test operations with the precursor **SPS**.



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# An "Apollo Pace" SPS Scenario

SPS-1982

BOEING

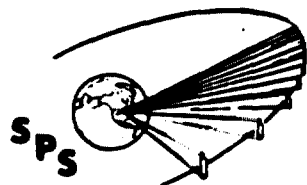
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|-----------------------------------------------|----|---------------------------------------|----|----|----|-------------------------------|----|----|----|---------------------------------------------|----|----|----|-----------------------------|----|----|----|------------------------|----|----|----|---|
| 78                                            | 79 | 80                                    | 81 | 82 | 83 | 84                            | 85 | 86 | 87 | 88                                          | 89 | 90 | 91 | 92                          | 93 | 94 | 95 | 96                     | 97 | 98 | 99 | 0 |
| <div>SPS TECHNOLOGY<br/>VERIFICATION</div>    |    |                                       |    |    |    |                               |    |    |    |                                             |    |    |    |                             |    |    |    |                        |    |    |    |   |
| LAUNCH<br>VEHICLES                            |    | ▼<br>SHUTTLE<br>DERIVATIVE<br>ON-LINE |    |    |    | ▼<br>HEAVY<br>LIFT<br>ON-LINE |    |    |    | ▼<br>HLLV<br>414-FLTS/YEAR                  |    |    |    | ▼<br>HLLV<br>1243 FLTS/YEAR |    |    |    |                        |    |    |    |   |
| CONSTRUCTION<br>BASES                         |    | ▼<br>PRECURSOR<br>BASE ON-LINE        |    |    |    | ▼<br>BASE 1<br>ON-LINE        |    |    |    | ▼<br>BASE 2<br>ON-LINE                      |    |    |    | ▼<br>BASE 3<br>ON-LINE      |    |    |    | ▼<br>BASE 4<br>ON-LINE |    |    |    |   |
| SPS ON-LINE<br>(10 GW EACH, EXCEPT PRECURSOR) |    |                                       |    |    |    | PRECURSOR<br>▼                |    |    |    | 1/2 1 2 3 5 7 10 13 17<br>▼ ▼ ▼ ▼ ▼ ▼ ▼ ▼ ▼ |    |    |    |                             |    |    |    |                        |    |    |    |   |
| 78                                            | 79 | 80                                    | 81 | 82 | 83 | 84                            | 85 | 86 | 87 | 88                                          | 89 | 90 | 91 | 92                          | 93 | 94 | 95 | 96                     | 97 | 98 | 99 | 0 |

- NOTE:
- 1) THE PRECURSOR SPS SERVES AS A DEVELOPMENTAL TOOL RATHER THAN AS A "COMMITMENT PRODUCER."
  - 2) BY 2000, THE INSTALLED UNITS CAN BE PRODUCING APPROXIMATELY 80% OF THE ELECTRICAL ENERGY CONSUMED IN THE U.S. IN 1976.



**GROUND BASED TECHNOLOGY ADVANCEMENT PLAN**

Shown here are the principal technology advancement areas and a preliminary estimate of the annual funding required to pursue these areas. No significant differences in the technology advancement plan have been identified since the completion of Part II. The technology advancement plan is described in additional detail in Volume 2 of the Part II Final Report.



SPS-1989

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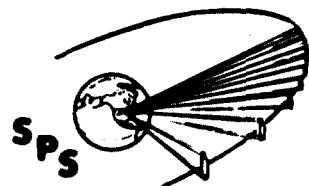
# Ground-Based Technology Advancement Plan

**BOEING**

| TECHNOLOGY AREA                     | YEARS | 1     | 2     | 3     | 4    | 5    | TOTAL |
|-------------------------------------|-------|-------|-------|-------|------|------|-------|
| SOLAR CELLS                         |       | 2.5   | 3.6   | 3.85  | 3.7  | 2.8  | 16.25 |
| THERMAL ENGINES & THERMAL SYSTEMS   |       | 2.5   | 3.5   | 3.75  | 3.5  | 2.5  | 15.75 |
| MICROWAVE POWER TRANSMISSION SYSTEM |       | 6.0   | 7.5   | 8.75  | 8.5  | 6.5  | 37.25 |
| SPACE STRUCTURES                    |       | 0.75  | 2.0   | 2.2   | 2.0  | 1.8  | 8.75  |
| MATERIALS                           |       | 1.5   | 2.0   | 2.5   | 2.0  | 2.0  | 10.0  |
| FLIGHT CONTROL SYSTEMS              |       | 0.5   | 0.8   | 1.0   | 0.9  | 0.8  | 4.0   |
| CONSTRUCTION SYSTEMS                |       | 3.0   | 4.0   | 4.5   | 5.5  | 5.5  | 22.5  |
| TRANSPORTATION SYSTEMS              |       | 4.5   | 7.5   | 8.75  | 7.5  | 7.5  | 35.75 |
| POWER DISTRIBUTION AND CONTROLS     |       | 1.5   | 2.0   | 2.5   | 3.5  | 2.5  | 12.0  |
| SPACE ENVIRONMENT FACTORS           |       | 2.0   | 2.8   | 2.45  | 2.2  | 2.0  | 11.25 |
| TOTALS                              |       | 24.75 | 35.50 | 40.25 | 39.3 | 33.7 | 173.5 |

**OVERALL TECHNOLOGY ADVANCEMENT PROGRAM**

The overall technology advancement program includes, in addition to the ground based program, shuttle spacelab sortie tests and a solar power technology advancement test article. The shuttle spacelab sortie test would include tests of beam fabricator machine and RF equipment and also tests of prototype manipulator systems and other construction aids. The solar power technology advancement test article would be a solar array in the 100 - 500 kilowatt range constructed and supported using the shuttle as an operating base. This test article would use solar arrays similar in nature to those planned for SPS. It would also use similar structural techniques and would provide a test base for testing of higher power RF equipment, electric thrusters and tests operating the solar array, at high generating voltages, to develop information on plasma interactions.

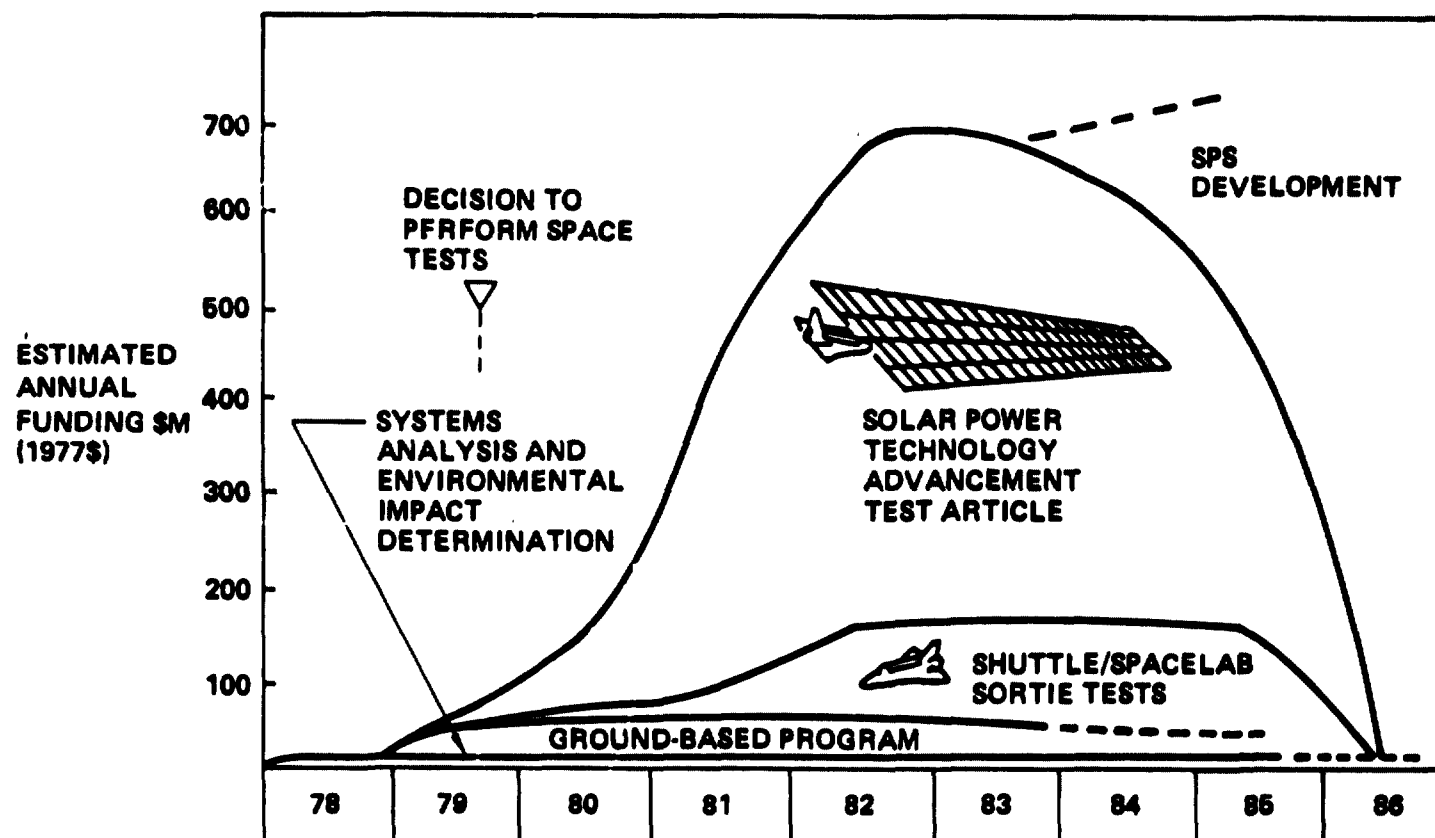


SPS-1988

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# Overall Technology Advancement Program

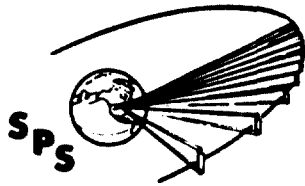
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### SIZE SENSITIVITY RESULTS POWER TRANSMISSION OPTIMIZATION

A size sensitivity design model was constructed and exercised. The first run of the model optimized power transmitter and rectenna sizes at the nominal power level of approximately 5,000 megawatts per link. The new results, although executed in somewhat more detail than earlier results, confirmed the earlier estimates that the optimum rectenna size is  $3/4$  of the transmitted beam diameter and that the optimum transmitter size is in the vicinity of 1.4 kilometers. However, transmitter sizes larger than one kilometer violate the peak beam intensity limit of 23 millowatts per centimeter squared. Therefore the best system uses a 1 kilometer transmitter and a rectenna diameter  $3/4$  of the beam diameter.

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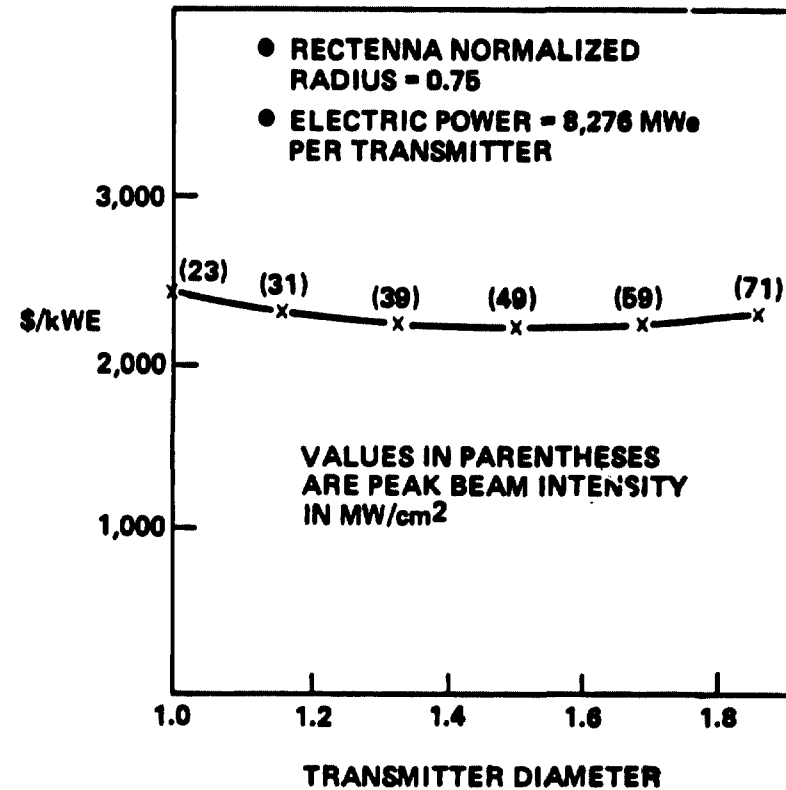
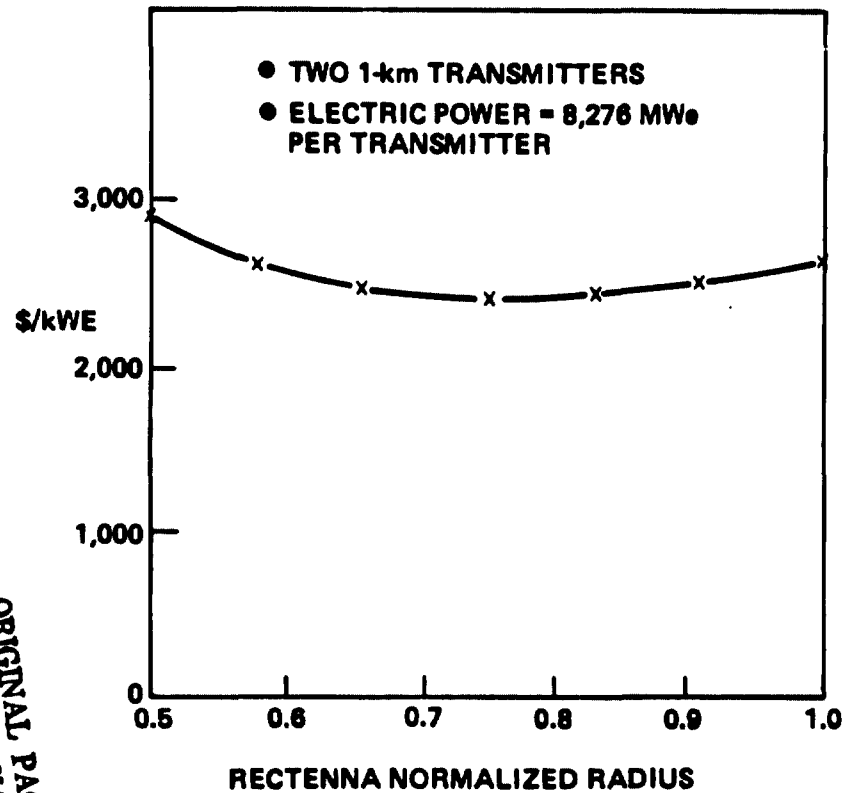


# Size Sensitivity Results Power Transmission Optimization

SPS-1980

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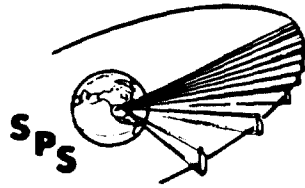
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**SIZE SENSITIVITY ANALYSIS MODELING DETAIL**

The size sensitivity model was implemented on the ISAI AH modeling system. The model consisted of 37 designer selected variables and 95 computed variables. The values generated by the model for the nominal design point of a 1 kilometer diameter transmitter are shown on the chart. A complete design point was generated for each sensitivity point analyzed.



# Size Sensitivity Model Details

SPS-1987

BOEING

| NOMINAL VALUES, INDEPENDENT VARIABLES |                     |
|---------------------------------------|---------------------|
| 1 SUMMER SOLSTICE FACTOR              | 9.7452E-01          |
| 2 COSINE LOSS                         | 9.1900E-01          |
| 3 SOLAR CELL EFFICIENCY               | 1.7300E-01          |
| 4 RADIATION DEGRADATION               | 9.7000E-01          |
| 5 TEMPERATURE DEGRADATION             | 9.9400E-01          |
| 6 COVER UV DEGRADATION                | 9.9600E-01          |
| 7 CELL-TO-CELL MISMATCH               | 9.9000E-01          |
| 8 CELL & PANEL LOST AREA              | 9.6100E-01          |
| 9 STRING I-SQ-R                       | 9.9800E-01          |
| 10 PANEL/BLANKET LOST AREA            | 9.8000E-01          |
| 11 DC-RF CONVERSION                   | 0.5000E-01          |
| 12 WAVEGUIDE I-SQ-R                   | 9.8500E-01          |
| 13 INTER-SUBARRAY ERRORS              | 9.4400E-01          |
| 14 INTRA-SUBARRAY ERRORS              | 9.8100E-01          |
| 15 ATMOS. ABSORPTION                  | 9.8000E-01          |
| 16 GRID INTERFACE                     | 9.7000E-01          |
| 17 POWER ACROSS ROTARY JOIN           | 8.2000E+03 MEGAWATT |
| 18 SIDELobe LIMIT                     | 1.0000E-01 MM/CM2   |
| 19 ANTENNA DIAMETER                   | 1.0000E+00 KM       |
| 20 RECTENNA NORMALIZED RAD            | 7.5000E-01          |
| 21 NOMINAL BAY SIZE                   | 6.6000E+02 METERS   |
| 22 SPS WIDTH                          | 8.0000E+00 DAYS     |
| 23 CLEARANCE IN BAY                   | 9.0000E+00 METERS   |
| 24 NO. OF ANTENNAS                    | 2.0000E+00          |
| 25 BEAM ELEMENT SIZE                  | 1.5000E+01 METERS   |
| 26 POWER PER KLYSTRON                 | 7.2000E+01 KWH      |
| 27 SUBARRAY SIZE                      | 1.0400E+01 METERS   |
| 28 ATTITUDE CONTROL ISP               | 2.0000E+04 SEC      |
| 29 MASS GROWTH                        | 2.6000E+01 PCT      |
| 30 TRANSP & CONSTR TIME               | 1.4000E+00 YEARS    |
| 31 INTEREST RATE                      | 7.5000E+00 PCT      |
| 32 PLANT FACTOR                       | 9.2000E+01 PCT      |
| 33 KLYSTRON "A" VOLTAGE               | 4.0000E+04 VOLTS    |
| 34 KLYSTRON "B" VOLTAGE               | 3.8000E+04 VOLTS    |
| 35 RECTENNA LATITUDE                  | 3.5000E+01 DEGREES  |
| 36 "A" BUS PWR FRAC                   | 6.3000E-01          |
| 37 "B" BUS PWR FRAC                   | 3.7000E-01          |

## SOLUTION RESULTS

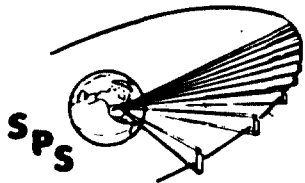
|                              |                    |                           |                    |
|------------------------------|--------------------|---------------------------|--------------------|
| 1 LIGHT INPUT EFFICIENCY     | 8.979E-01          | 51 PEAK ANT THERMAL PWR   | 4.750E+00 KW/M2    |
| 2 NET CELL EFFICIENCY        | 1.601E-01          | 52 DC OUTPUT POWER        | 4.860E+00 GW/LINE  |
| 3 BASIC CONVERSION EFFY      | 1.360E-01          | 53 GRID POWER             | 9.445E+00 GW TOTAL |
| 4 BLANKET FACTORS            | 1.399E-01          | 54 LAND AREA PER SECT     | 1.790E+00 M2       |
| 5 BUS I-SQ-R                 | 1.415E-01          | 55 "Y" MOM OF INERTIA     | 1.161E+14 KG-M2    |
| 6 NET ENERGY CONV EFFY       | 1.203E-01          | 56 THRUST PER CORNER      | 8.840E+01 NEWTONS  |
| 7 AREAWISE EFFICIENCY        | 9.700E-01          | 57 NUMBER OF THRUSTERS    | 8.840E+01 PER INST |
| 8 ANTENNA POWER DISTR EFFY   | 9.701E-01          | 58 CONTROL POWER          | 1.111E+02 MEGAWATT |
| 9 NET DC-RF EFFICIENCY       | 8.123E-01          | 59 ANNUAL PROPELLANT      | 3.625E+01 TONS     |
| 10 IDEAL BEAM EFFICIENCY     | 9.650E-01          | 60 STRUCTURE MASS         | 5.700E+03 TONS     |
| 11 NET BEAM EFFICIENCY       | 8.955E-01          | 61 CONTROL SYS MASS       | 1.976E+02 TONS     |
| 12 INTERCEPT EFFICIENCY      | 9.512E-01          | 62 SOLAR BLANKET MASS     | 4.344E+04 TONS     |
| 13 RECTENNA RF-DC EFFICIENCY | 8.670E-01          | 63 POWER DISTR MASS       | 2.431E+03 TONS     |
| 14 NET RF LINK EFFY          | 8.340E-01          | 64 MECH & ELEC R/J MASS   | 1.022E+02 TONS     |
| 15 DC-TO-DC EFFICIENCY       | 5.679E-01          | 65 ANT STRUC MASS         | 5.000E+02 TONS     |
| 16 DC-TO-GRID EFFICIENCY     | 5.702E-01          | 66 ANT WAVEGUIDE MASS     | 4.314E+03 TONS     |
| 17 OVERALL PHYSICAL EFFY     | 4.861E-02          | 67 ANT KLYSTRON MASS      | 1.310E+04 TONS     |
| 18 AREA EFFECTIVE EFFY       | 6.420E-02          | 68 ANT CONTROL CKTS MASS  | 1.024E+03 TONS     |
| 19 BLANKET AREA              | 1.017E+00 M2       | 69 ANT PWR DISTR MASS     | 9.471E+02 TONS     |
| 20 ANTENNA DIA               | 1.000E+00 KM       | 70 ANT PWR PROCATC MASS   | 4.881E+03 TONS     |
| 21 REQUIRED SIDELobe SUPPR   | 2.363E+01 DB       | 71 ANT MASS               | 2.485E+04 TONS     |
| 22 TAPER REQUIRED FOR SL BU  | 9.555E+00 DB       | 72 STRUCTURE COST         | 2.850E+01 BILLION  |
| 23 TRANSMITTER POWER TAPER   | 1.000E+01 DB       | 73 CONTROL SYS COST       | 8.893E+02 BILLION  |
| 24 RECEIVER AVG/PEAK RATIO   | 2.061E-01          | 74 SOLAR BLANKET COST     | 3.560E+00 BILLION  |
| 25 XMR AVG/PEAK RATIO        | 3.909E-01          | 75 POWER DISTR COST       | 6.321E+02 BILLION  |
| 26 BEAM SPREAD FACTOR        | 1.458E+00          | 76 MECH/ELEC R/J COST     | 2.146E+02 BILLION  |
| 27 RADIATED RF POWER         | 6.725E+03 MEGAWATT | 77 ANT STRUC COST         | 3.485E+01 BILLION  |
| 28 BEAM DIAMETER             | 1.270E+01 KM       | 78 ANT WAVEGUIDE COST     | 2.508E+01 BILLION  |
| 29 BEAM AREA                 | 1.267E+00 M2       | 79 ANT KLYSTRON COST      | 5.997E+01 BILLION  |
| 30 AVERAGE BEAM POWER DENS   | 4.753E+00 KW/CM2   | 80 ANT CONTROL CKTS COST  | 2.261E+01 BILLION  |
| 31 PEAK BEAM INTENSITY       | 2.506E+01 KW/CM2   | 81 ANT PWR DISTR COST     | 1.023E+01 BILLION  |
| 32 POWER IN MAIN BEAM        | 6.023E+03 MEGAWATT | 82 ANT PWR PROCATC COST   | 3.560E+01 BILLION  |
| 33 SATELLITE LENGTH          | 3.104E+01 DAYS     | 83 ANT COST               | 1.872E+00 BILLION  |
| 34 NUMBER OF DAYS            | 2.484E+02 DAYS     | 84 NO OF FREIGHT FLIGHTS  | 5.563E+02          |
| 35 XMR PWR DISTR LOSS        | 2.985E-02          | 85 CREW SERVICE NO OF FLT | 1.933E+01          |
| 36 ADJ BAY USEFUL AREA       | 4.896E+05 M2       | 86 DTS COST               | 9.077E+01 BILLION  |
| 37 BAY SIZE                  | 4.800E+02 METERS   | 87 TOTAL TRANSP COST      | 6.709E+00 BILLION  |
| 38 SPS AREA                  | 1.006E+12 KM2      | 88 RECTENNA COST          | 4.749E+00 BILLION  |
| 39 MEAN SOLAR INSOLATION     | 1.469E+02 GW       | 89 CONSTRUCTION COST      | 1.160E+00 BILLION  |
| 40 SOLAR CELL OUTPUT         | 1.759E+01 GW       | 90 INTEREST DURING CONSTR | 1.417E+00 BILLION  |
| 41 ROTARY JOINT CURRENT "A"  | 1.344E+05 AMPS     | 91 LATITUDE AREA FACTOR   | 1.419E+00          |
| 42 ROTARY JOINT CURRENT "B"  | 7.895E+04 AMPS     | 92 TOTAL MASS             | 9.666E+04 TONS     |
| 43 TOTAL PROCESSED POWER     | 2.484E+03 MEGAWATT | 93 TOTAL COST             | 2.268E+01 BILLION  |
| 44 TOTAL KLYSTRON INPUT      | 1.607E+04 MEGAWATT | 94 COST/KWH               | 2.402E+03 \$       |
| 45 TOTAL KLYSTRON OUTPUT     | 1.366E+04 MEGAWATT | 95 COST/KWH               | 4.464E+01 MILLS    |
| 46 NUMBER OF KLYSTRONS       | 1.897E+03          |                           |                    |
| 47 MAX KLYSTRON PACKING DEN  | 3.291E+01 PER SUB  |                           |                    |
| 48 MAX RF POWER DENSITY      | 2.191E+01 KW/M2    |                           |                    |
| 49 NUMBER OF SUBARRAYS       | 7.261E+03 PER ANT  |                           |                    |
| 50 RECTENNA AREA             | 7.671E+07 M2       |                           |                    |

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### SIZE SENSITIVITY ANALYSIS POWER LEVEL AND TRANSMITTER DIAMETER

This chart shows a joint optimization of transmitter diameter and power level holding the rectenna size constant at the optimum value. As the system power level is reduced it is possible to employ somewhat larger transmitting antennas without violating the  $23 \text{ mw/cm}^2$  limit. Transmitter diameters larger than 1.4 kilometers do not pay off; the minimum system cost in dollars per kilowatt follows along the  $23 \text{ mw/cm}^2$  limit to about 2500 megawatts and then follows up the 1.4 kilometer diameter transmitter curve. Note that comparatively little cost penalty is incurred going down as low as 3000 megawatts of grid power. Below 3,000 megawatts the system cost in dollars per kilowatt begins to turn up rapidly.

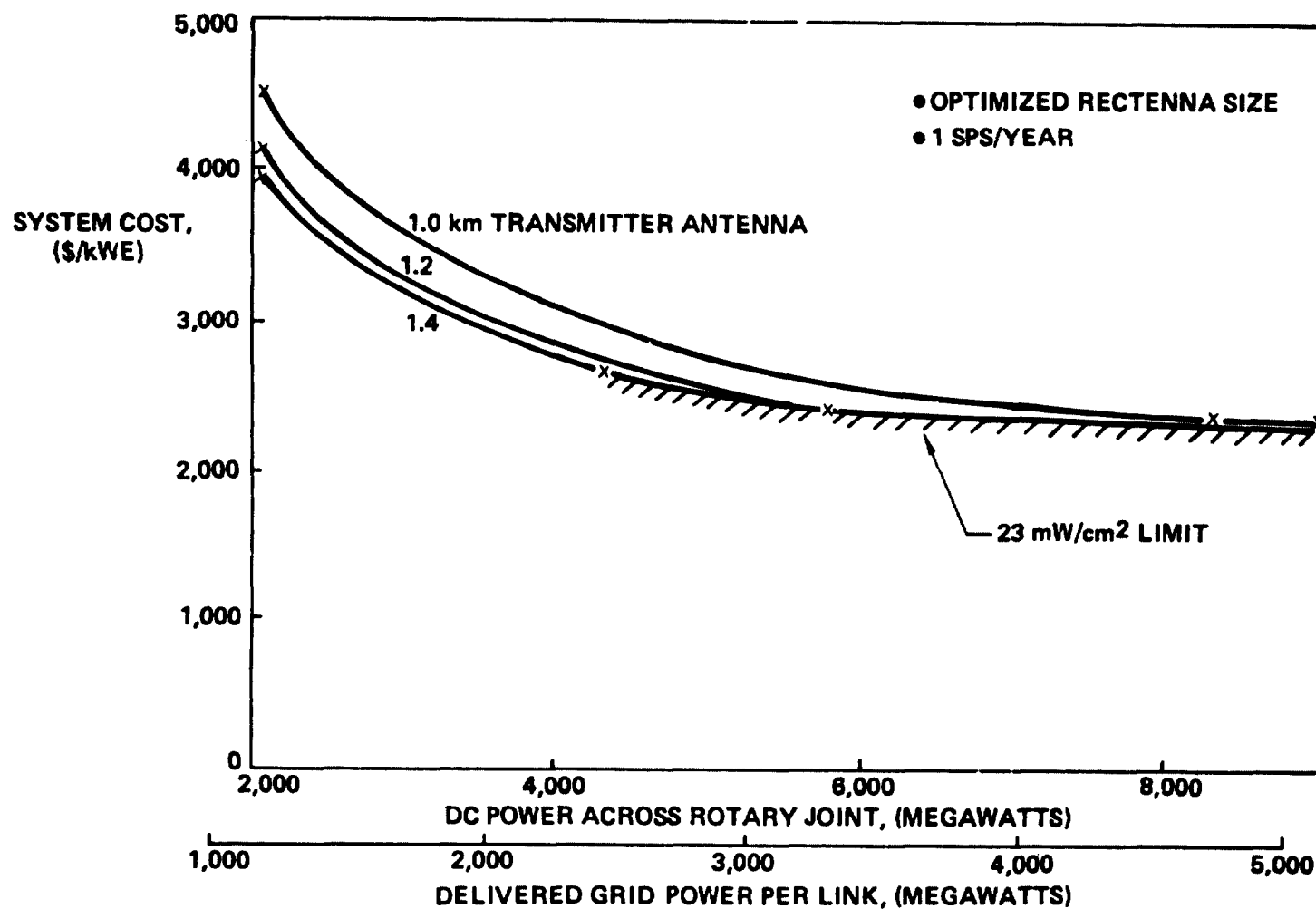


SPS-1991

D180-24071-3

# Size Sensitivity Analysis Power Level and Transmitter Diameter

BOEING

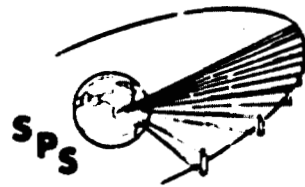


### LEO TRANSPORTATION

The Earth to LEO transportation effort, during this portion of the study, concentrated on definition of 1) a 2-stage fully reusable winged SPS cargo vehicle and 2) a 2-stage reusable ballistic recoverable concept. A number of significant changes have been incorporated into the SPS cargo vehicle since the completion of the Part II study effort. These changes on the cargo vehicle include:

- o A methane/liquid oxygen fueled booster with flyback capability
- o Delta winged stages with a crew manning the orbiter and also the capability in the orbiter to transport personnel
- o Incorporation of a mid-body cargo bay in the orbiter capable of handling a payload density of  $75 \text{ kg/m}^3$

The 2-stage ballistic recoverable vehicle has a payload in the 90 metric ton class that could possibly support the SPS precursor program.



D180-24071-3

# Leo Transportation

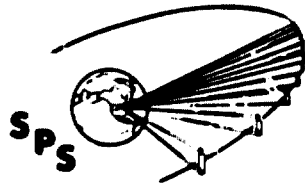
**BOEING**

- TWO-STAGE FULLY REUSABLE WINGED FREIGHTER
- TWO-STAGE BALLISTIC RECOVERABLE CANDIDATE CONCEPT FOR THE PRECURSOR PROGRAM
- SUMMARY

**POTENTIAL LAUNCH VEHICLE**

The two vehicle concepts studied, identified by the shading on the opposite chart, are highlighted in the potential launch vehicle candidate family. The net payload capability to low earth orbit is

- ≈ 400 000 kg for the 2-stage winged SPS cargo vehicle
- ≈ 90 000 kg for the 2-stage ballistic recoverable vehicle

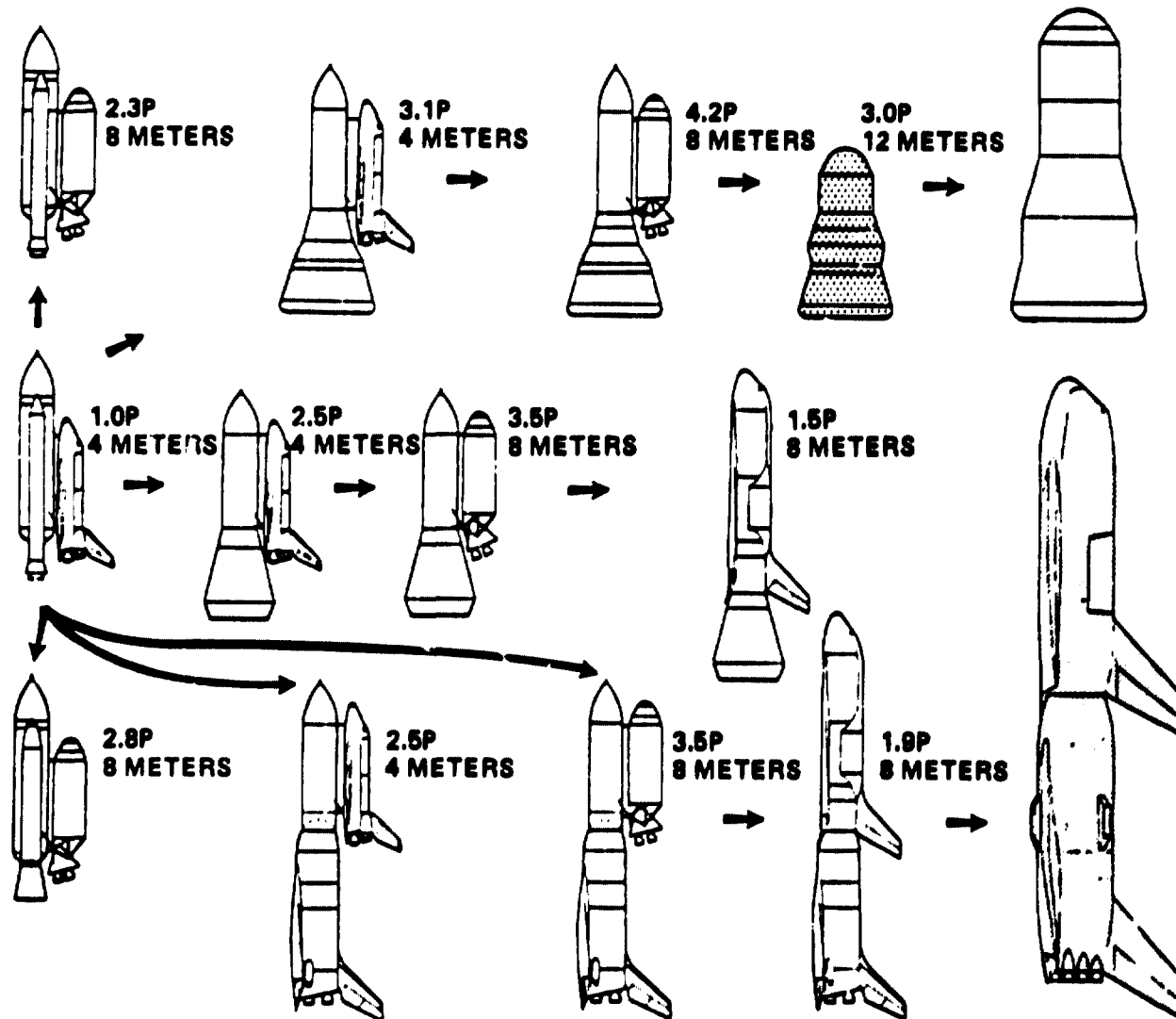


D180-24071-3

## Potential Launch Vehicles

SPS-1967

BOEING



**TWO-STAGE WINGED SPS LAUNCH VEHICLE  
(FULLY REUSABLE CARGO CARRIER)**

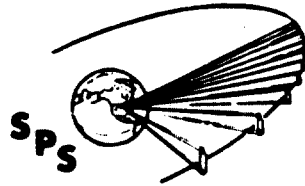
The launch configuration of the SPS cargo vehicle is shown on the adjacent chart with the overall geometry noted. This series burn concept uses 16  $\text{LCH}_4/\text{LO}_2$  engines on the booster and 14 standard SSME's on the orbiter. The  $\text{LCH}_4/\text{LO}_2$  booster engines are a gas generator cycle providing a vacuum thrust of  $9.79 \times 10^6$  newtons each. The SSME's on the orbiter provide a vacuum thrust of  $2.09 \times 10^6$  newtons (100% power level). The nominal 100% power level for the SSME's was selected based on engine life considerations which indicated about a 3 factor reduction in life if the 109% power level is used.

An airbreather propulsion system (12 installations of an SST type engine) has been provided on the booster for flyback capability to simplify the booster operational mode. The reference wing for both stages is

$$\begin{aligned} S_W (\text{Orbiter}) &= 1446 \text{ m}^2 \quad (15,560 \text{ ft}^2) \\ S_W (\text{Booster}) &= 2330 \text{ m}^2 \quad (25,080 \text{ ft}^2) \end{aligned}$$

Heat sink thermal protection system is provided on the booster and the Shuttle's Reusable Surface Insulation (RSI) is used on the orbiter.

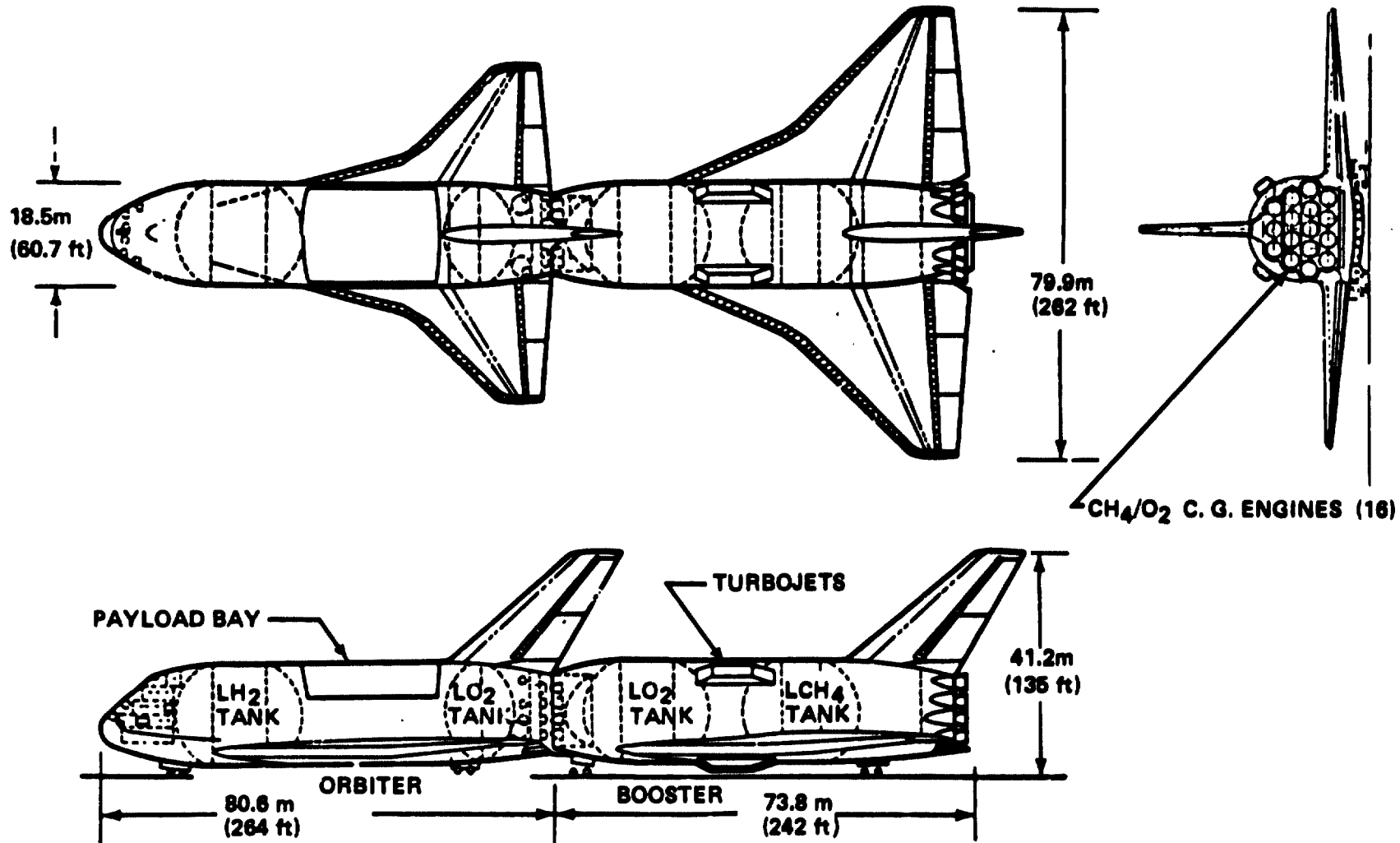
D180-24071-3



## Two-Stage Winged SPS Launch Vehicle (Fully Reusable Cargo Carrier)

SPS-1953

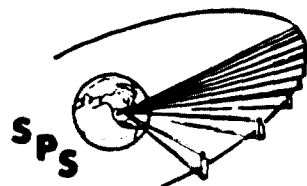
BOEING





## **2-5. AGE WINGED VEHICLE DESIGN CHARACTERISTICS**

The vehicle characteristics are noted on the adjacent chart. The net delivered payload is 424 000 kg. A return payload of 15% (63 500 kg) of the delivered payload was assumed for the orbiter entry and landing conditions. The resulting mass fraction is 0.875 for the booster and 0.841 for the orbiter.



D180-24071-3

## Two-Stage Winged Vehicle Design Characteristics

SPS-1962

BOEING

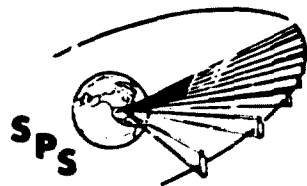
|                                      | ORBITER   |            | BOOSTER   |
|--------------------------------------|-----------|------------|-----------|
| GLOW                                 |           | 10,978,400 |           |
| BLOW                                 | —         |            | 7,813,700 |
| BOOSTER FUEL (LCH <sub>4</sub> )     | —         |            | 1,708,900 |
| BOOSTER OXIDIZER (LO <sub>2</sub> )  | —         |            | 5,126,700 |
| BOOSTER INERTS                       | —         |            | 978,100   |
| OLOW—LESS PAYLOAD                    | 2,740,700 |            | —         |
| ORBITIER FUEL (LH <sub>2</sub> )     | 329,400   | }          | —         |
| ORIBITER OXIDIZER (LO <sub>2</sub> ) | 1,976,200 |            | —         |
| ORBITER INERTS                       | 435,100   |            | —         |
| ASCENT PAYLOAD                       | 424,000   |            | —         |
| RETURN PAYLOAD ~ 15%                 | 63,500    |            | —         |
| MASS FRACTION                        | 0.841     |            | 0.875     |
| ENTRY WEIGHT—NO PAYLOAD              | 335,200   |            | 936,600   |
| —WITH RETURN P/L                     | 456,000   |            | —         |
| START CRUISE WEIGHT—NO P/L           | —         |            | 932,900   |
| —WITH RETURN P/L                     | —         |            | —         |
| LANDING WEIGHT—NO PAYLOAD            | 391,800   |            | 846,700   |
| —WITH RETURN P/L                     | 452,600   |            | —         |

(ALL MASS DATA IN kg)

\* MAINSTAGE + FLIGHT PERFORMANCE RESERVE

**ASCENT PERFORMANCE CHARACTERISTICS**

The SP5 launch vehicle ascent characteristics are noted on the adjacent chart. A '3g' maximum acceleration thrust profile was used due to the manned capability and also to minimize the load conditions on the orbiter. The booster staging velocity of 2170 m/sec is well within the "heat sink" capability of the aluminum/titanium airframe.



SPS-1951

D180-1 071-3

## Ascent Performance Characteristics

**BOEING**

### FIRST STAGE

|                             |   |            |             |
|-----------------------------|---|------------|-------------|
| T/W AT IGNITION             | - | 1.30       |             |
| MAXIMUM DYNAMIC PRESSURE    | - | 35.91 kPa  | (750 psf)   |
| MAXIMUM ACCELERATION        | - | 3.0g       |             |
| STAGE BURN TIME             | - | 155.24 sec |             |
| RELATIVE STAGING VELOCITY   | - | 2170 m/sec | (7,120 fps) |
| DYNAMIC PRESSURE AT STAGING | - | 1.16 kPa   | (24 psf)    |

### SECOND STAGE

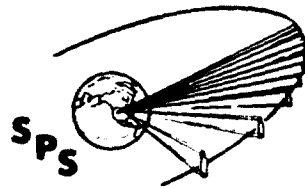
|                      |   |            |  |
|----------------------|---|------------|--|
| INITIAL T/W          | - | 0.94       |  |
| MAXIMUM ACCELERATION | - | 3.0 g      |  |
| STAGE BURN TIME      | - | 350.24 sec |  |

**D180-24071-3**

**S/S WINGED VEHICLE  
REENTRY CHARACTERISTICS**

The reentry characteristics for the booster and orbiter are noted on the opposite chart. The maximum deceleration for the booster is 4.27 g's and the subsonic transition altitude is 17.86 km. The orbiter reentry has been limited to a normal load factor of 1.41 g's until the subsonic transition which occurs at an altitude of 13.62 km.

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6rS-1954

# SPS Winged Vehicle Reentry Characteristics

BOEING

## BOOSTER

### APOGEE CONDITIONS

$h = 80.82 \text{ km}$

$V_{rel} = 1955 \text{ m/sec}$

### MAXIMUM DECELERATION CONDITION

$q = 10.77 \text{ kPa}$

$h = 32.61 \text{ km}$

$V_{rel} = 1327 \text{ m/sec}$

NORMAL LOAD FACTOR = 4.27 g's

### MAXIMUM DYNAMIC PRESSURE CONDITION

$q = 13.29 \text{ kPa}$

$h = 22.96 \text{ km}$

$V_{rel} = 686 \text{ m/sec}$

NORMAL LOAD FACTOR = 1.49 g's

### SUBSONIC TRANSITION CONDITION

$h = 17.86 \text{ km}$

$\alpha = 15 \text{ deg}$

## ORBITER

### MAXIMUM DYNAMIC PRESSURE CONDITION

$q = 13.17 \text{ kPa}$

$h = 15.55 \text{ km}$

$V_{rel} = 361 \text{ m/sec}$

NORMAL LOAD FACTOR = 1.41

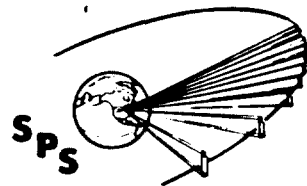
### SUBSONIC TRANSITION CONDITION

$h = 13.62 \text{ km}$

$\alpha = 6.4 \text{ deg}$

**SPS BOOSTER MASS STATEMENT**

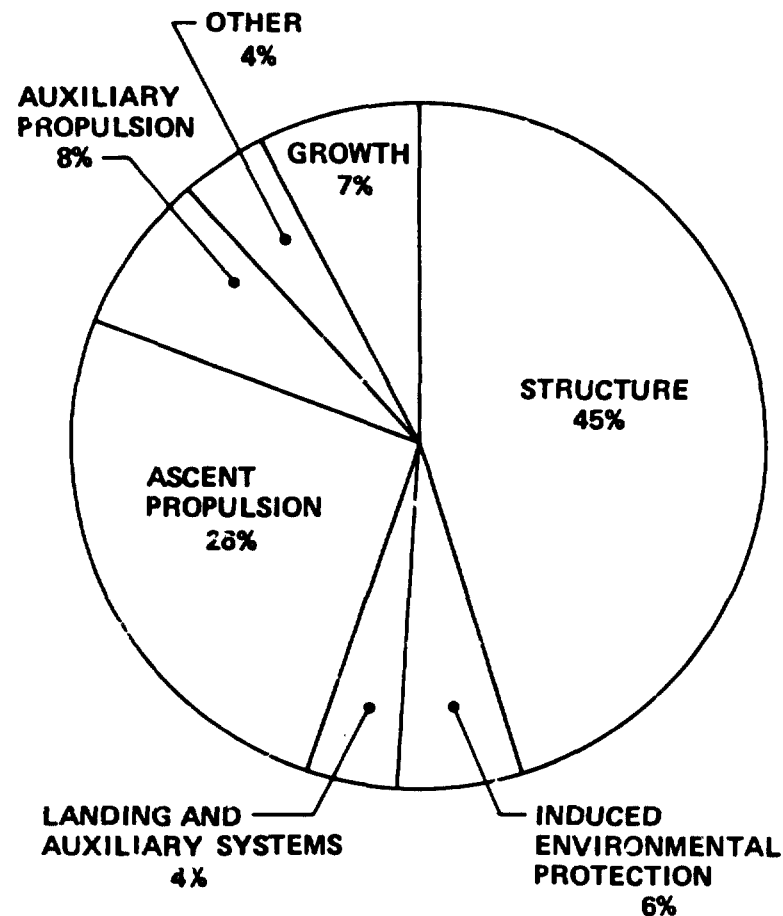
The flyback booster mass characteristics are shown on the opposite chart. The structure, induced environment protection, ascent and auxiliary propulsion, and landing subsystems account for 89% of the dry mass. The induced environmental protection subsystem mass includes the additional structural thickness required for "heat sink capability" and the base heat shield.



# Booster Mass Statement

SPS-1956

BOEING



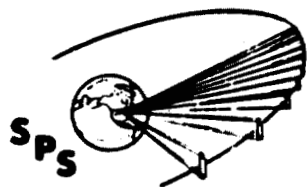
DRY MASS BREAKDOWN

|                                        | MASS (kg)      |
|----------------------------------------|----------------|
| STRUCTURE                              | 360 800        |
| INDUCED ENVIRONMENTAL PROTECTION       | 46 400         |
| LANDING AND AUXILIARY SYSTEMS          | 34 500         |
| ASCENT PROPULSION                      | 204 600        |
| AUXILIARY PROPULSION                   | 60 600         |
| PRIME POWER                            | 4 300          |
| ELECTRICAL CONVERSION AND DISTRIBUTION | 4 200          |
| HYDRAULIC CONVERSION AND DISTRIBUTION  | 10 900         |
| SURFACE CONTROLS                       | 10 300         |
| AVIONICS                               | 1 500          |
| ENVIRONMENTAL CONTROL                  | 200            |
| GROWTH                                 | 58 600         |
| <b>DRY MASS =</b>                      | <b>796 900</b> |
| RESIDUALS AND RESERVES                 | 49 800         |
| <b>LANDING MASS =</b>                  | <b>846 700</b> |
| LOSSES DURING FLYBACK                  | 86 200         |
| <b>START FLYBACK MASS =</b>            | <b>932 900</b> |
| ENTRY IN-FLIGHT LOSSES                 | 3 700          |
| <b>START ENTRY MASS =</b>              | <b>936 600</b> |
| IN-FLIGHT LOSSES PRIOR TO ENTRY        | 27 000         |
| <b>STAGING MASS =</b>                  | <b>963 600</b> |
| THRUST DECAY PROPELLANT                | 14 500         |
| <b>INERT MASS =</b>                    | <b>978 100</b> |



**SPS ORBITER MASS STATEMENT**

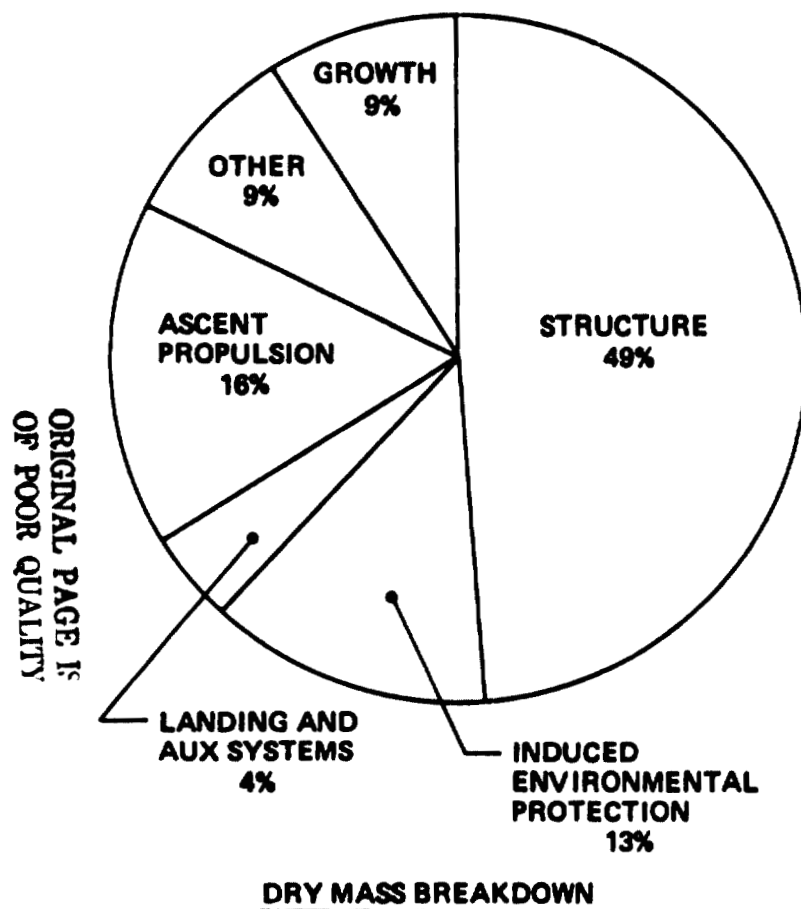
The orbiter mass characteristics are shown on the opposite chart. Structure accounts for approximately 50% of the stage dry mass. The ascent propulsion and thermal protection subsystems are an additional 29% of the dry mass. The dry mass is 86% of the inert mass with the remainder including residuals and reserves, personnel and payload accommodations, and inflight losses.



SPS-1955

BOEING

# Orbiter Mass Statement



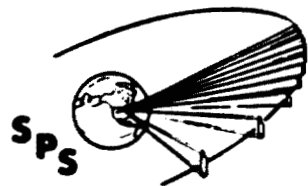
|                                        | MAS (kg)       |
|----------------------------------------|----------------|
| STRUCTURE                              | 182 900        |
| INDUCED ENVIRONMENTAL PROTECTION       | 48 300         |
| LANDING AND AUX SYSTEMS                | 15 800         |
| ASCENT PROPULSION                      | 60 800         |
| AUXILIARY PROPULSION                   | 9 500          |
| PRIME POWER                            | 2 500          |
| ELECTRICAL CONVERSION AND DISTRIBUTION | 4 800          |
| HYDRAULIC CONVERSION AND DISTRIBUTION  | 3 600          |
| SURFACE CONTROLS                       | 6 800          |
| AVIONICS                               | 2 400          |
| ECLSS AND PERSONNEL PROV               | 2 900          |
| GROWTH                                 | 32 800         |
| <b>DRY MASS -</b>                      | <b>373 200</b> |
| PERSONNEL AND PAYLOAD ACCOMMODATIONS   | 4 100          |
| RESIDUAL AND RESERVES                  | 14 500         |
| <b>LANDING MASS -</b>                  | <b>391 800</b> |
| ENTRY IN-FLIGHT LOSSES                 | 3 400          |
| <b>START ENTRY MASS -</b>              | <b>395 200</b> |
| IN-FLIGHT LOSSES PRIOR TO ENTRY        | 39 900         |
| <b>INERT MASS -</b>                    | <b>435 100</b> |

### **BOOSTER PROCESSING TIMELINES**

The booster timeline from launch to its move to the integration position is shown on the chart. These timelines reflect the average turnaround times for the mature system. A total of 62 hours is estimated for this portion of the turnaround with the scheduled and unscheduled maintenance activity requiring 36 hours. On-board condition monitoring equipment will enhance the operations by

- 1) Providing performance monitoring of the subsystems
- 2) Aiding in fault isolation and detection

Rocket engine maintenance is anticipated to be the major portion of the booster operations.

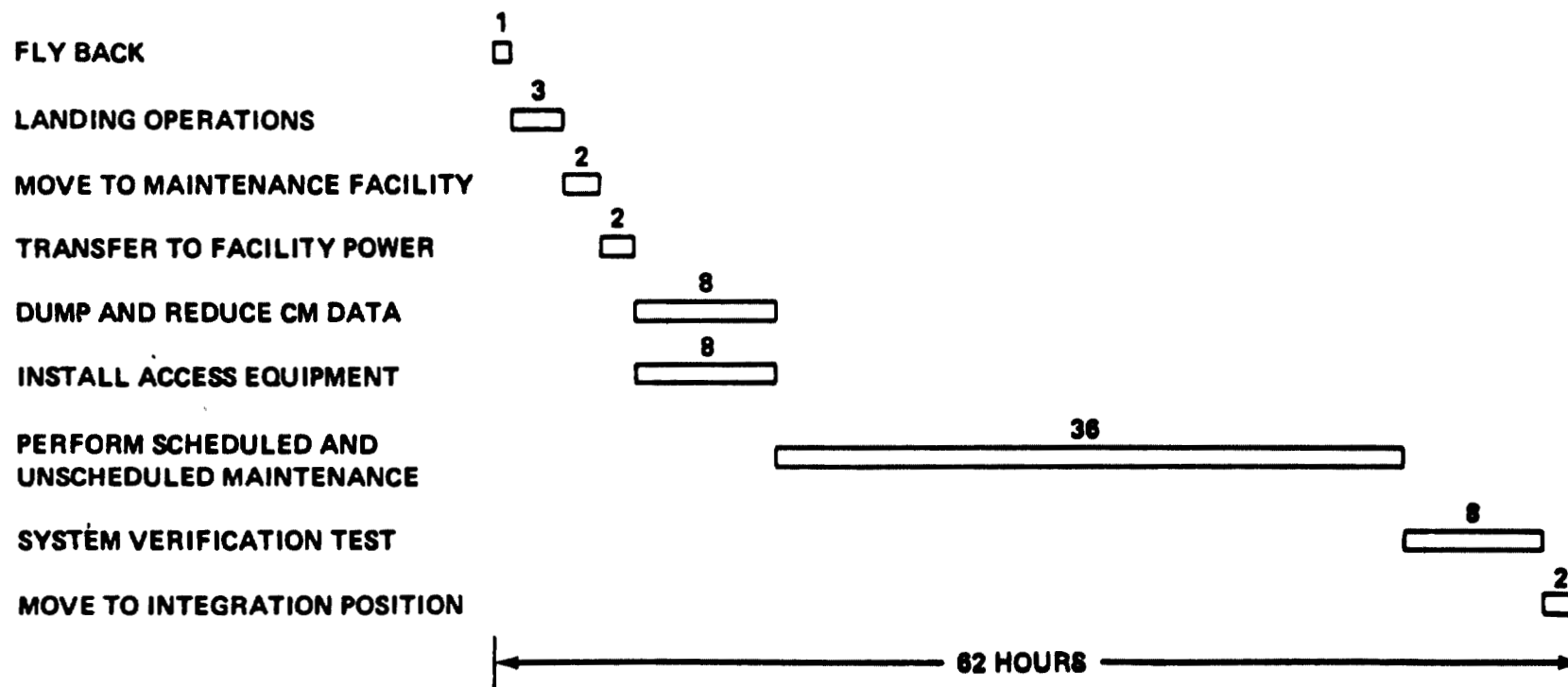


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## Booster Processing Timelines

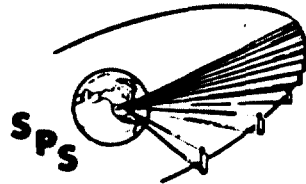
SPS-1961

**BOEING**



**ORBITER PROCESSING TIMELINES**

The orbiter timeline from launch to its move to the integration position is shown on the chart. A total time of 97 hours for orbiter processing including the 24 hour-on-orbit staytime is estimated for the mature system. The maintenance activity is estimated to be 48 hours, due to the thermal protection system and the additional systems/equipment required for the manned stage. A total of 12 hours has been allocated for payload installation in a parallel operation with the orbiter maintenance.



D180-24071-3

# Orbiter Processing Timelines

SPS-1960

BOEING

ON-ORBIT STAY TIME  
AND DEORBIT

LANDING OPERATIONS

MOVE TO MAINTENANCE FACILITY

TRANSFER TO FACILITY POWER

DUMP AND REDUCE CM DATA

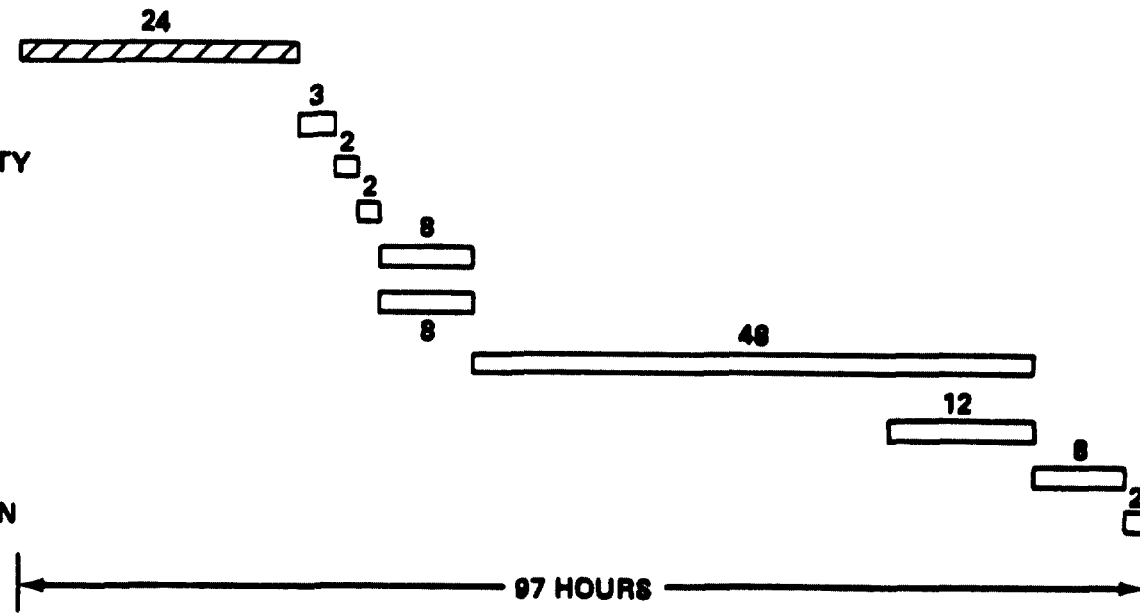
INSTALL ACCESS EQUIPMENT

PERFORM SCHEDULED AND  
UNSCHEDULED MAINTENANCE

INSTALL PAYLOAD

SYSTEM VERIFICATION TEST

MOVE TO INTEGRATION POSITION

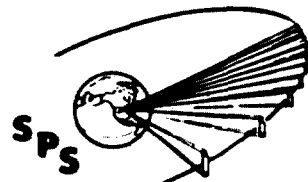


**D180-24071-3**

### **INTERGRATED VEHICLE OPERATIONS TIMELINES**

The integrated vehicle timelines for operations beginning with booster positioning through launch are shown on the adjacent chart. This portion of the launch operations requires 34 hours for the booster and 30 hours for the orbiter.

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D180-24071-3

# Integrated Vehicle Operations Timelines

SPS-1962

BOEING

INSTALL 1ST STAGE  
ON LAUNCHER/ERECTOR

4

INSTALL 2ND STAGE  
ON LAUNCHER/ERECTOR

4

INSTALL ORDNANCE AND  
CLOSE OUT

3

PERFORM VEHICLE INTEGRATION TEST

4

ROTATE TO VERTICAL

4

RETRACT INTERMEDIATE  
SUPPORTS

2

MAKE INTERFACE CONNECTIONS  
AND CONDUCT PRELAUNCH  
VERIFICATION

4

FUEL LCH<sub>4</sub>, LO<sub>2</sub>, LH<sub>2</sub>

2 3.5 2.5

COUNTDOWN AND LAUNCH

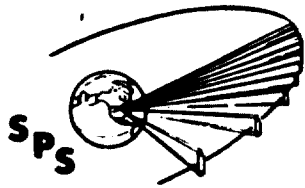
1

34 HOURS



**VEHICLE TURNAROUND ANALYSIS SUMMARY**

The total turnaround times for the SPS winged launch vehicle are shown on the chart. The total booster turnaround time is 97 hours and the corresponding orbiter time is 127 hours. The estimated 2-stage ballistic recoverable turnaround times are shown for reference. The flyback capability on the booster, with its inherent return to launch site ability, provides for a minimum turnaround time.



D180-24071-3

# Vehicle Turnaround Analysis Summary

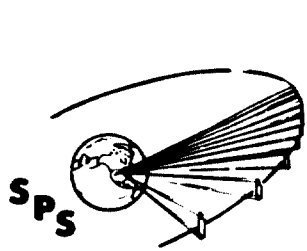
SPS-1957

**BOEING**

| VEHICLE CONCEPT                               | STAGE OPS ONLY        | INTEGRATION AND LAUNCH OPS | TOTAL TURNAROUND       |
|-----------------------------------------------|-----------------------|----------------------------|------------------------|
| WING/WING<br>BOOSTER<br>ORBITER               | 63 HOURS<br>97 HOURS  | 34 HOURS<br>30 HOURS       | 97 HOURS<br>127 HOURS  |
| BALLISTIC/BALLISTIC<br>BOOSTER<br>UPPER STAGE | 93 HOURS<br>102 HOURS | 34 HOURS<br>30 HOURS       | 127 HOURS<br>132 HOURS |

**SPS LAUNCH VEHICLE DDT&E COST**

The DDT&E cost per the flight hardware and its associated ground support equipment is shown on the adjacent chart for both the booster and orbiter stages. The total development cost for both stages is \$11.2B. Systems test, which includes all the ground and flight test hardware in addition to the test labor, accounts for in excess of 50% of the total development cost. The booster DDT&E cost includes a new rocket engine and airbreather engine development. The orbiter DDT&E reflects use of the Space Shuttle's SSME's and some of the other subsystems which were modified rather than new developments.

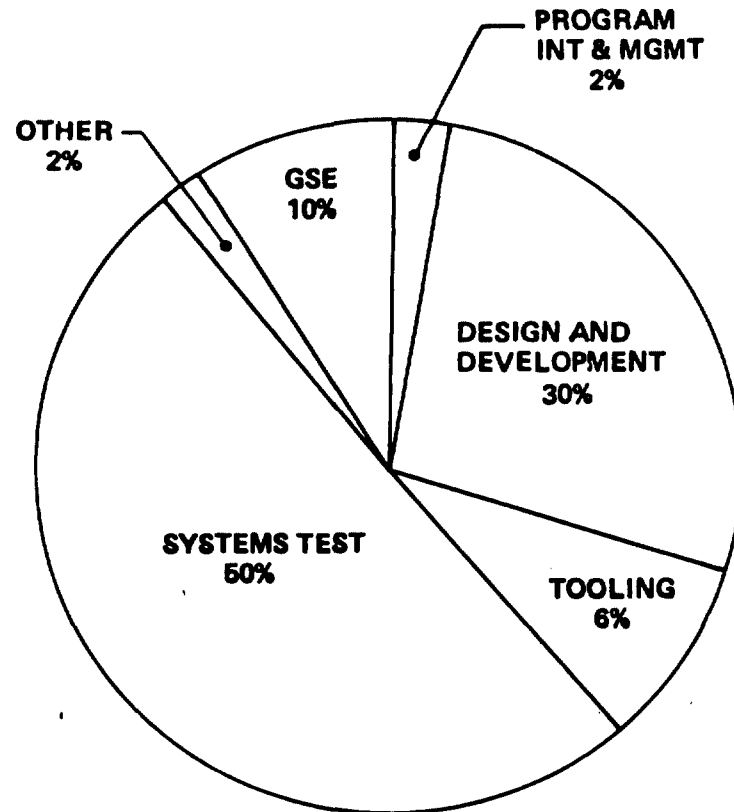


D180-24071-3

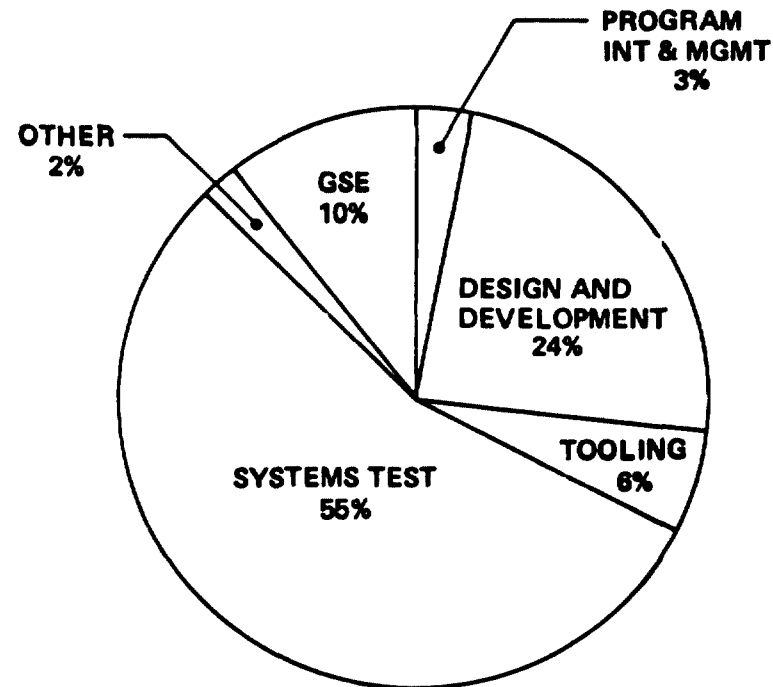
# SPS Vehicle DDT&E Cost

SPS-1959

BOEING



BOOSTER DDT&E = \$6,528M



ORBITER DDT&E = \$4,674M

● TOTAL VEHICLE DDT&E = \$11.2B (LESS FACILITIES)

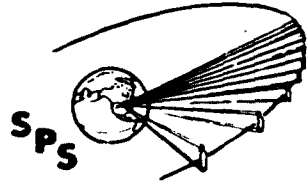
### SPS LAUNCH VEHICLE PRODUCTION COST

The initial unit production cost for both the SPS cargo vehicle booster and orbiter is shown on the adjacent chart. The theoretical first unit cost (TFU) for the booster of \$821.4M and \$638.5M for the orbiter were developed using the Boeing Parametric Cost Model (PCM). The following is a breakdown of the TFU cost by major subsystem:

| <u>SUBSYSTEM</u>      | <u>BOOSTER</u> | <u>ORBITER</u> |
|-----------------------|----------------|----------------|
| Structure             | 21%            | 16%            |
| TPS                   | N/A            | 10%            |
| Main Propulsion       | 24%            | 28%            |
| Landing and Aux. Sys. | 13%            | 9%             |
| Flyback Propulsion    | 11%            | N/A            |
| Other Subsystems      | 19%            | 26%            |

The ground support equipment TFU cost is estimated to be \$162.8M and \$126.0M for the booster and orbiter respectively.

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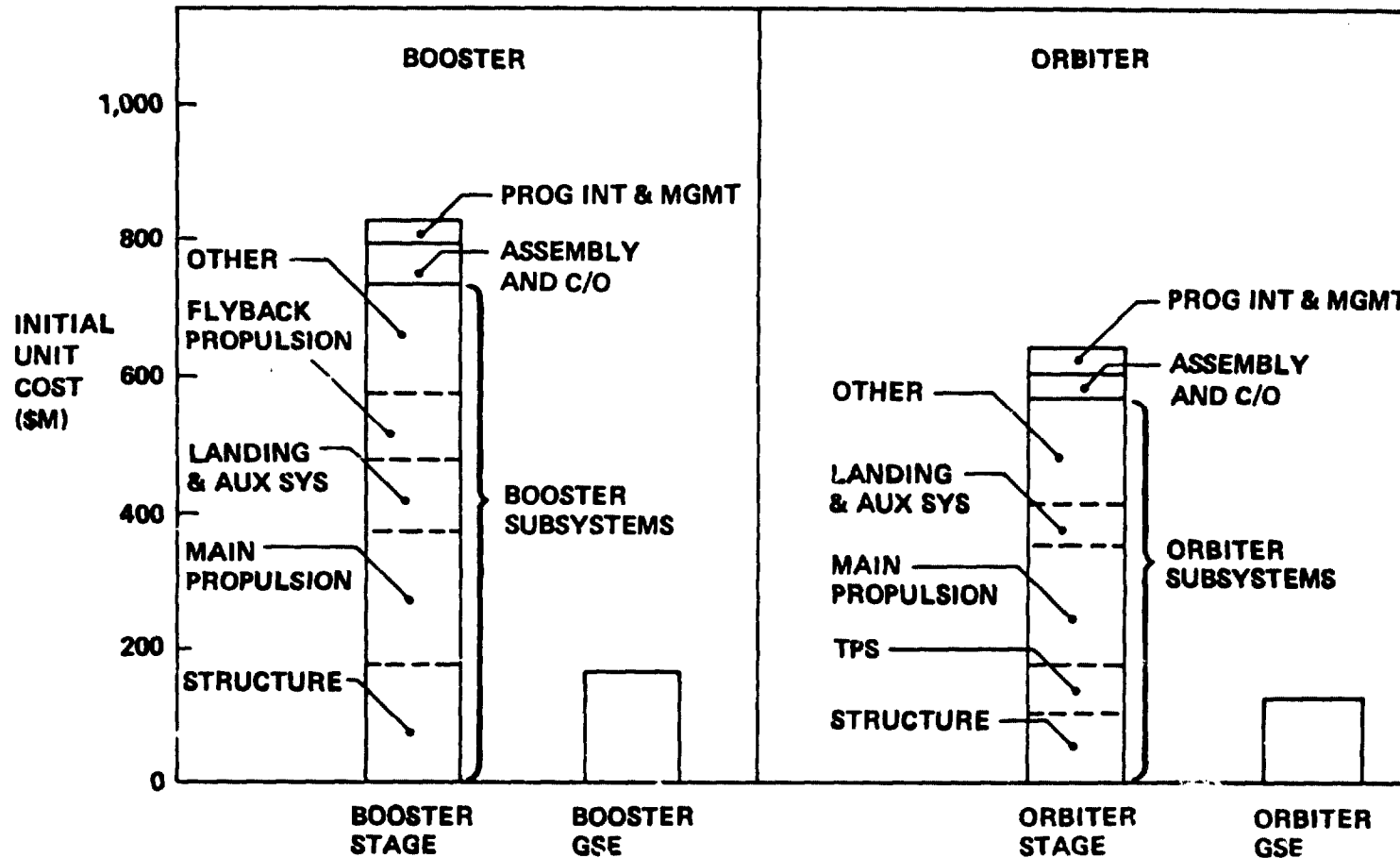


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# SPS Launch Vehicle Production Cost

SPS-1968

BOEING

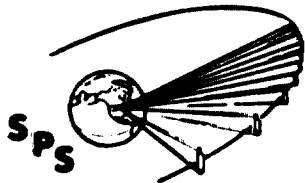


**SPS LAUNCH VEHICLE AVERAGE COST/FLIGHT  
(1 SATELLITE/YEAR)**

The cost/flight breakdown shown on the opposite page is the average for the 400 per year launch rate and 14 years of operation. The cost/flight items follow the Shuttle User Charge Policy guidelines with the following additions

- 1) Amortization of the fleet production costs
- 2) Inclusion of the rate tooling cost due to the hardware quantities required.

Flight Hardware production and spares is the largest single item with the booster and orbiter accounting for 55% and 45%, respectively. Propellant cost amounts to 12% of the total per flight cost.

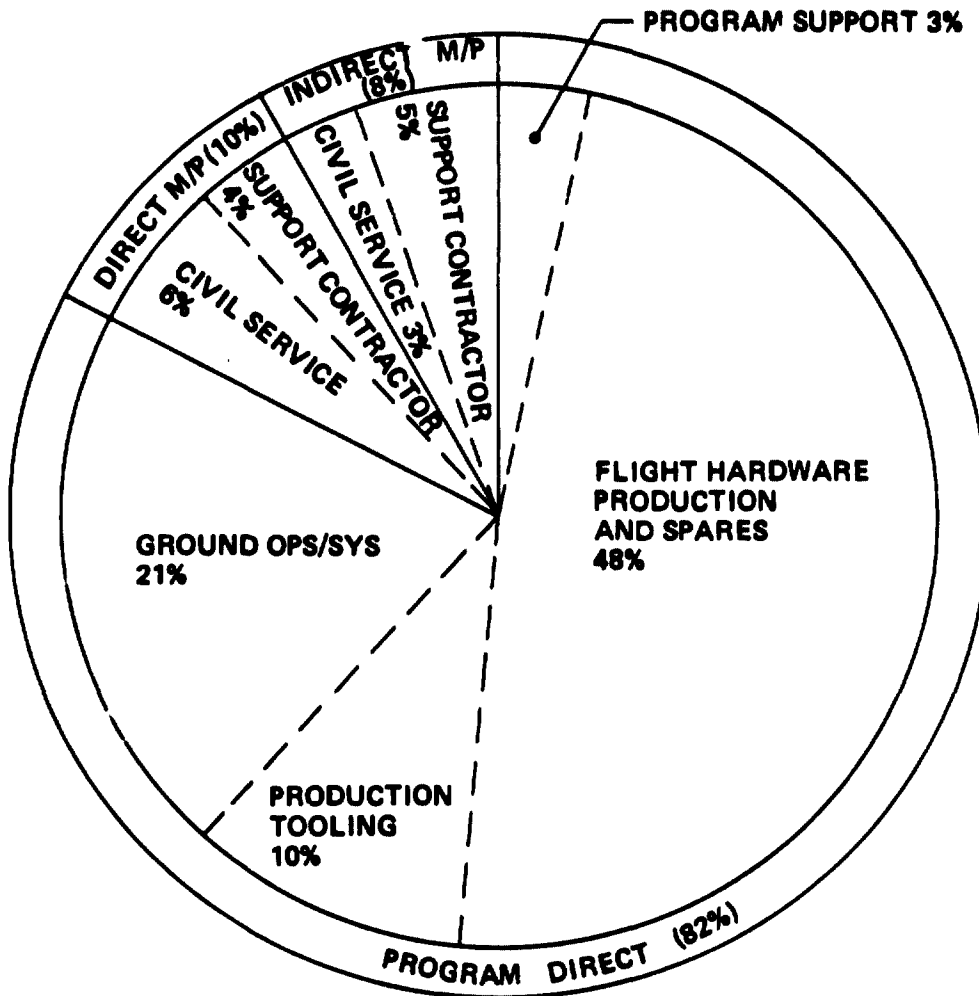


SPS-1983

D180-24071-3

# SPS Launch Vehicle Average Cost/Flight (One Satellite/Year)

**BOEING**



- TWO-STAGE WINGED VEHICLE
- PLACEMENT OF 1 SATELLITE PER YEAR (400 FLIGHTS)
- 14 YEAR PROGRAM
- NO ATTRITION

AVERAGE COST/FLIGHT = \$13.447M

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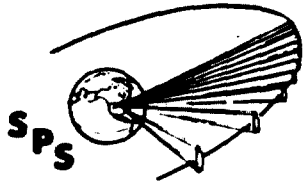


## EFFECT OF LAUNCH RATE ON COST PER FLIGHT

The chart on the opposite page illustrates the effect of launch rate on the average cost/flight and the transport cost to low Earth orbit for the SPS cargo vehicle. The required launch rate of approximately 400 flights per satellites results in the following:

| Annual Launch Rate | Cost/Flight | Transport Cost |          |
|--------------------|-------------|----------------|----------|
|                    |             | \$/kg          | (\$/lbm) |
| 400 Flights        | \$13.447M   | 31.71          | (14.38)  |
| 1600 Flights       | \$10.754M   | 25.36          | (11.50)  |

A 40 launch per year rate, comparable to the planned rate for Shuttle from KSC, would result in an average cost of \$23M per flight for the SPS cargo launch vehicle. Also noted on the chart, are the NASA/JSC in-house cost estimates as of January 1978.

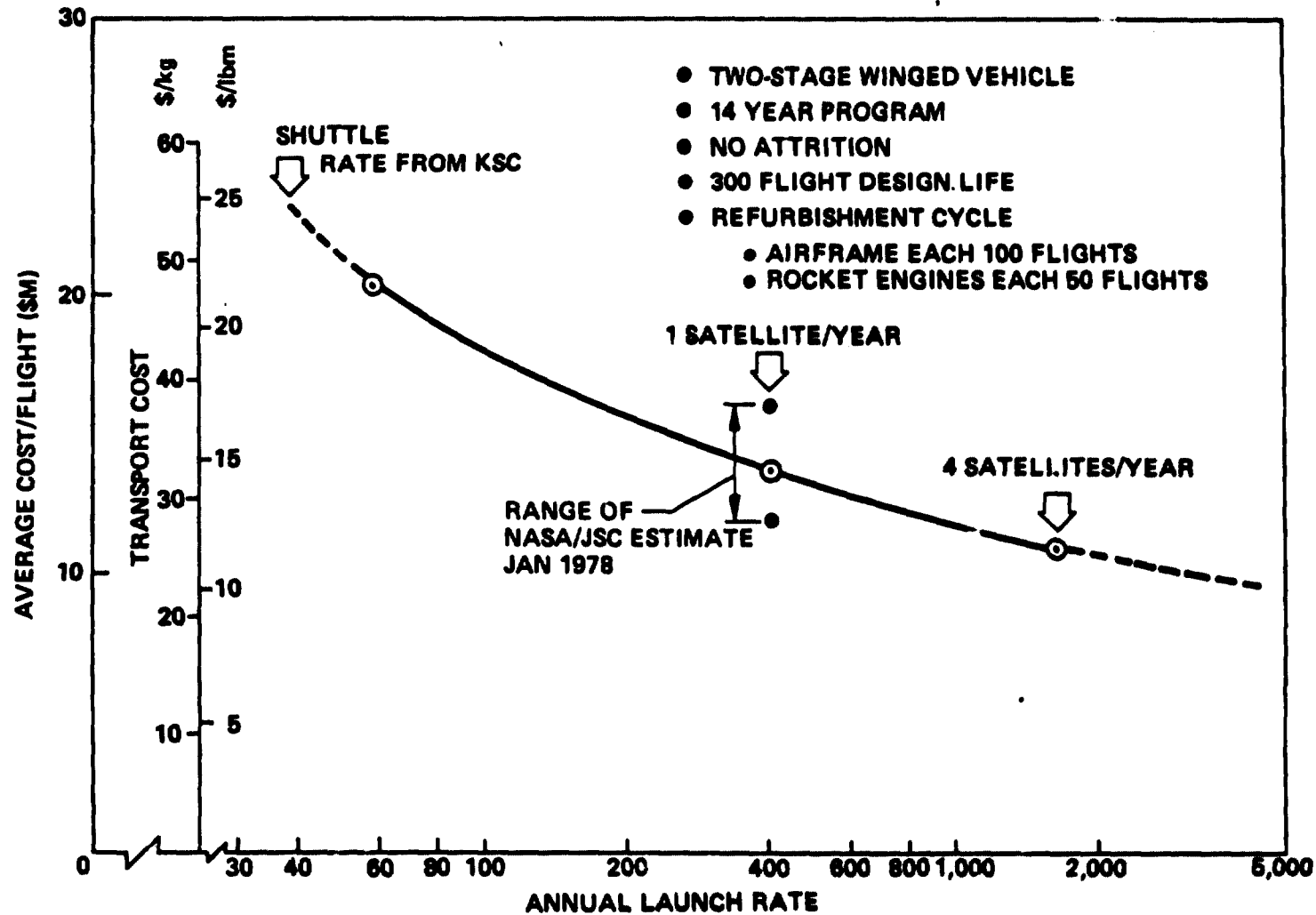


D180-24071-3

## Effect of Launch Rate on Cost Per Flight

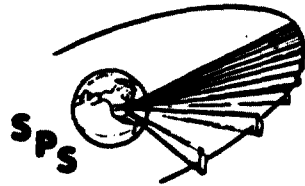
SPS-1964

BOEING



**EFFECT OF DESIGN LIFE AND ATTRITION RATE**

A sensitivity analysis was conducted to determine the impact of various design life and attrition rate criteria. The results of this analysis are shown on the adjacent chart. Attrition rates of between 0.1% and 1% were evaluated along with a design life criteria of 300 and 500 flights. The attrition rate influence for the range of values investigated resulted in a 32% variation in the average cost per flight. Design life has a decreasing influence as the attrition rate increases. A recommended criteria for the 2-stage winged vehicle is a 500 flight design life and 0.1% attrition rate which should be achievable within the time span available.

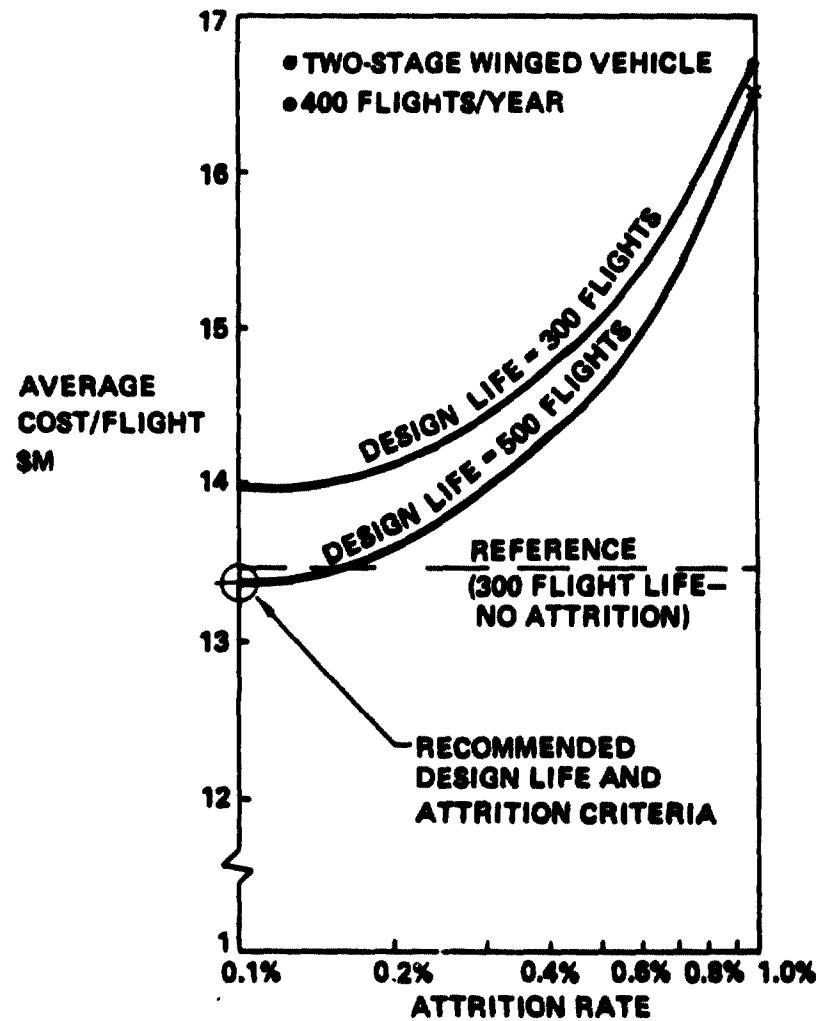


SPS-1966

D180-24071-3

## Effect of Design Life and Attrition Rate

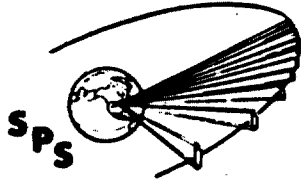
BOEING



### VEHICLE SIZE COMPARISON

A comparison of the physical size (planform area) for the SPS launch vehicle orbiter (2nd stage only), a single stage to orbit (SSTO) vehicle, and a 747 Freighter are shown on the opposite chart. For the two space transportation concepts, the gross liftoff weight (GLOW) to payload ratio are noted. The two-stage SPS vehicle has a liftoff weight to payload ratio of 25.9 which uses the benefits of staging, whereas the SSTO has a ratio of 19.1. The inert and dry mass data are noted on the chart. The "Advanced SSTO" mass data reflects the maximum to achieve the desired performance level. The required maximum SSTO mass is about 33% greater than the dry weight of a 747 Freighter or 47% of the SPS Orbiter (2nd Stage) inert mass.

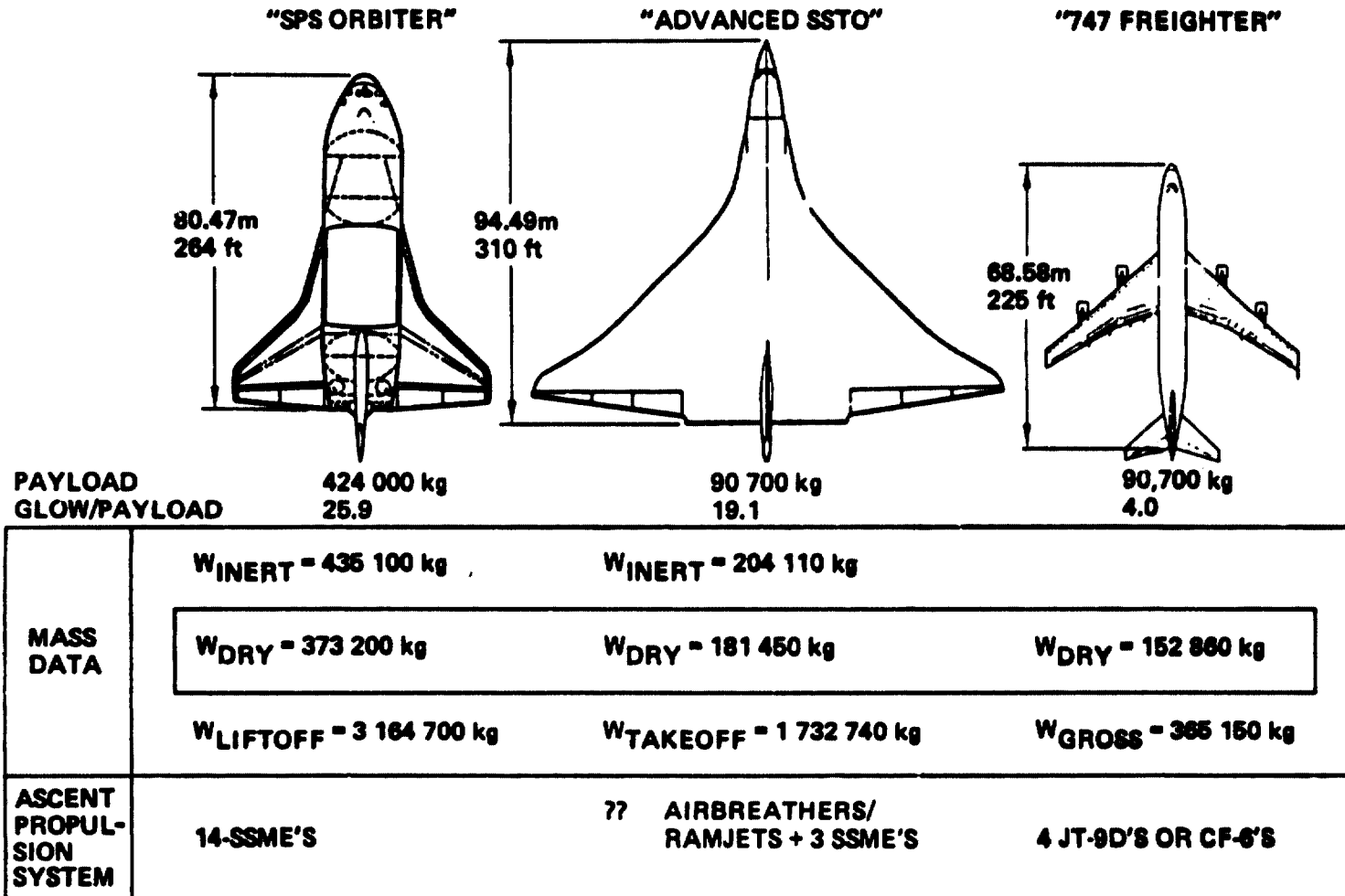
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SPS-1989

BOEING

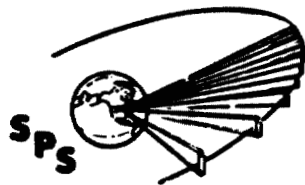
# Vehicle Size Comparison



### POTENTIAL LAUNCH VEHICLES

The other vehicle concept investigated in this portion of the SPS study is the 2-stage ballistic recoverable vehicle in the 90 metric ton payload class. The potential evolution of this vehicle is noted on the adjacent chart of launch vehicle family concepts. The vehicle evaluation would begin with a booster (including a new gas generator engine) to support an increased performance Space Shuttle, and then proceed to a fully reusable upper stage for cargo missions. The following charts will describe the 2-stage reusable ballistic concept.

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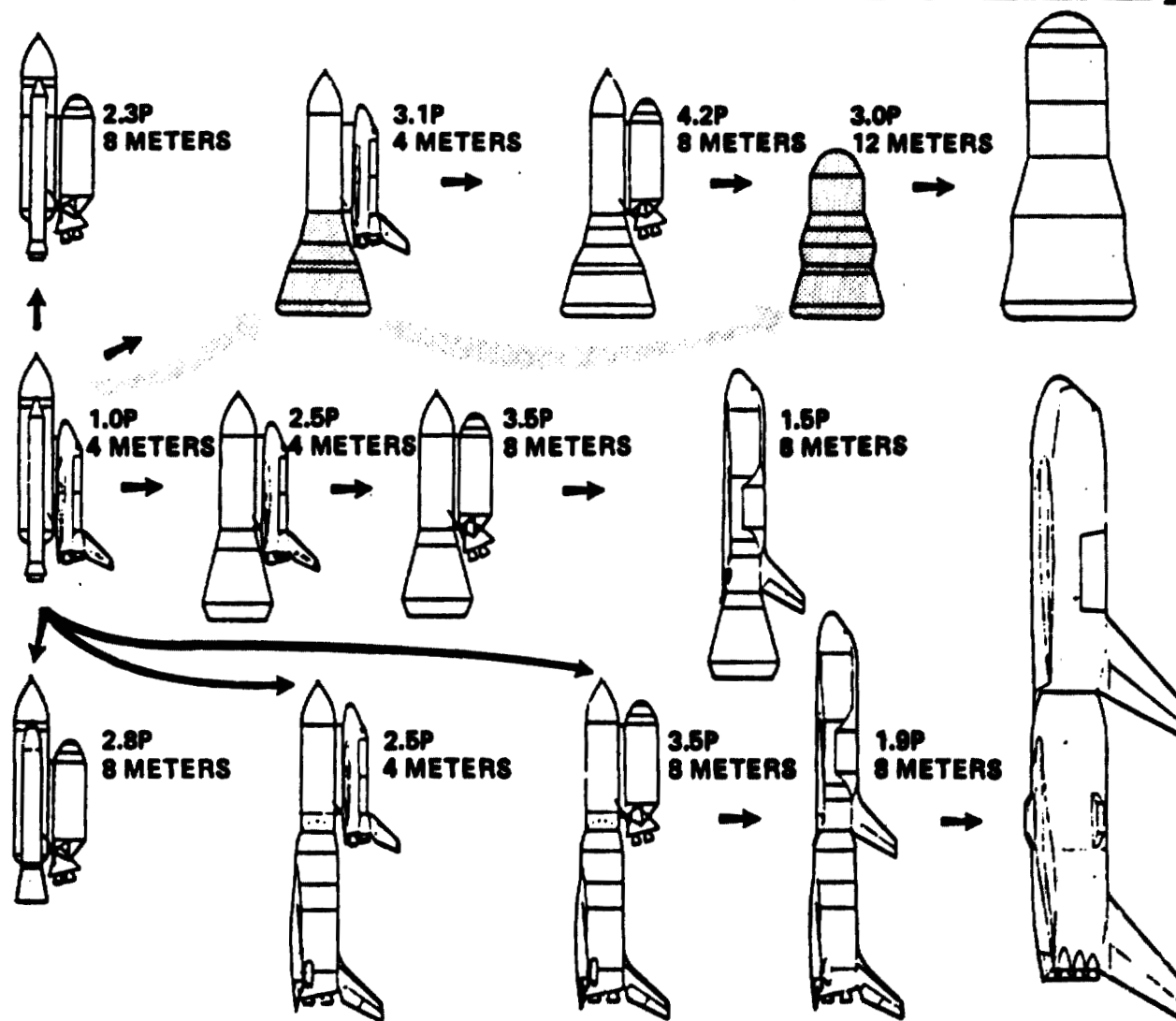


DI80-24071-3

# Potential Launch Vehicles

SPS-1987

BOEING



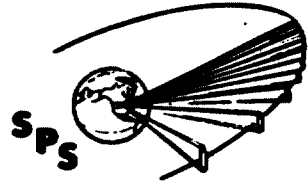


**TWO-STAGE BALLISTIC SPS LAUNCH VEHICLE  
(FULLY REUSABLE)**

The ballistic launch vehicle configuration is shown on the adjacent chart. This is a series burn configuration in which both stages re-enter ballistically and soft-land on the water. Ascent propulsion for the booster consists of 4  $\text{LCH}_4/\text{LO}_2$  gas generator cycle engines providing  $8.90 \times 10^6$  newtons vacuum thrust each. Four standard SSME's operating at 100% power level ( $2.09 \times 10^6$  newtons vacuum thrust each) provide ascent propulsion for the second stage.

Both stages are equipped with pressure fed landing engines to provide terminal deceleration after ballistic re-entry. Thermal protection for both ascent and re-entry is accomplished by water cooled base heat shields in both stages.

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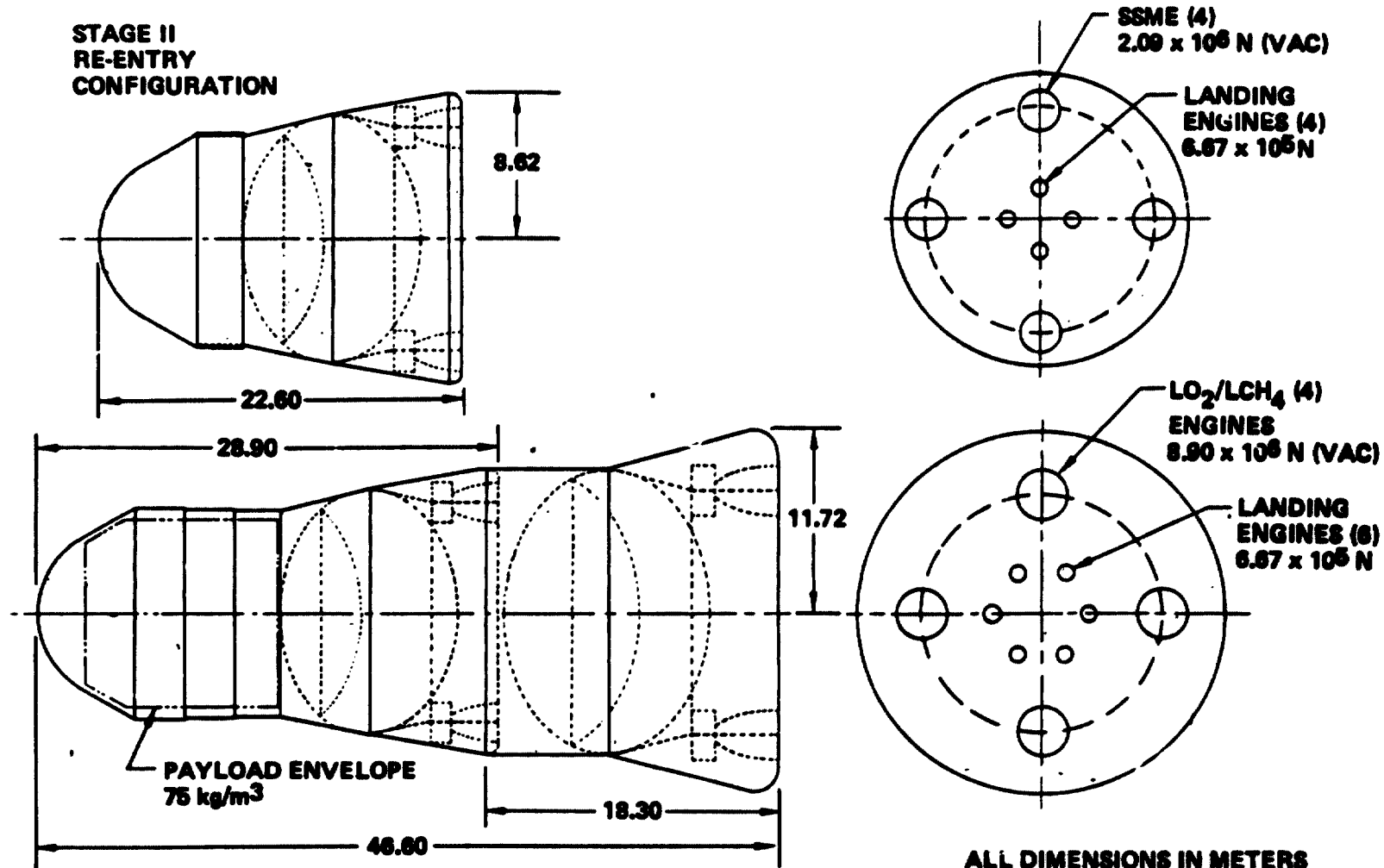


## Two-Stage Ballistic Vehicle Concept (Precursor)

SPS-1988

BOEING

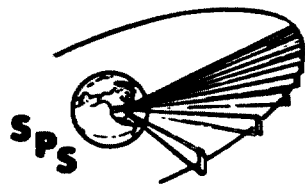
**STAGE II  
RE-ENTRY  
CONFIGURATION**



**D180-24071-3**

**TWO-STAGED BALLISTIC LAUNCH VEHICLE  
DESIGN CHARACTERISTICS**

The two-stage ballistic launch vehicle characteristics are specified on the adjacent chart. The vehicle delivers a net payload of 93,700 kg. The booster and orbiter stages have mass fractions of .903 and .812 respectively.



D180-24071-3

## 2 Stage Ballistic Vehicle (Precursor) Design Characteristics

SPS-1972

BOEING

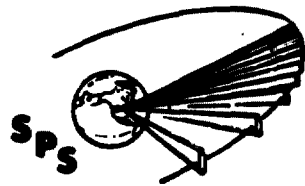
|                                     | <u>ORBITER</u> | <u>BOOSTER</u> |
|-------------------------------------|----------------|----------------|
| GLOW                                | 2,589,480      |                |
| BLOW                                | —              | 1,921,810      |
| BOOSTER FUEL (LCH <sub>4</sub> )    | —              | 434,000        |
| BOOSTER OXIDIZER (LO <sub>2</sub> ) | —              | 1,301,940      |
| BOOSTER INERTS                      | —              | 185,870        |
| LOW-LESS PAYLOAD                    | 573,970        | —              |
| ORBITER FUEL (LH <sub>2</sub> )     | 66,560         | —              |
| ORBITER OXIDIZER (LO <sub>2</sub> ) | 399,380        | —              |
| ORBITER INERTS                      | 108,020        | —              |
| ASCENT PAYLOAD                      | 93,700         | —              |
| MASS FRACTION                       | .812           | .73            |
| ENTRY WEIGHT                        | 99,500         | 180,190        |
| LANDING WEIGHT                      | 89,870         | 163,180        |

\*MAINSTAGE + FLIGHT PERFORMANCE  
RESERVE

(ALL MASS DATA IN kg)

**ASCENT PERFORMANCE CHARACTERISTICS**

The ascent performance characteristics of the two-stage ballistic vehicle are noted on the adjacent chart. The booster staging velocity is 2477 m/sec (8125 fps) and the maximum acceleration experienced is 4.23 g's at booster burnout.



SPS-1971

D180-24071-3

## SPS Ballistic Vehicle (Precursor) Ascent Performance Characteristics

BOEING

### FIRST STAGE

|                             |                         |
|-----------------------------|-------------------------|
| T/W @ IGNITION              | = 1.28                  |
| MAXIMUM DYNAMIC PRESSURE    | = 29.35 kPa (613 PSF)   |
| MAXIMUM ACCELERATION        | = 4.23 g's              |
| STAGE BURN TIME             | = 168.39 SEC            |
| RELATIVE STAGING VELOCITY   | = 2477 M/SEC (8125 FPS) |
| DYNAMIC PRESSURE AT STAGING | = 1.48 kPa (31 PSF)     |

### SECOND STAGE

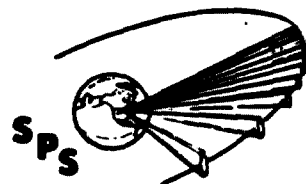
|                      |              |
|----------------------|--------------|
| INITIAL T/W          | = 1.28       |
| MAXIMUM ACCELERATION | = 4.15 g's   |
| STAGE BURN TIME      | = 246.84 SEC |

**D180-24071-3**

**SPS BALLISTIC LAUNCH VEHICLE  
MASS STATEMENT**

The mass characteristics of the ballistic launch vehicle are shown on the adjacent chart. Structure accounts for 46% and 59% of the dry masses of the booster and orbiter respectively. Ascent propulsion is the other major fraction of booster and orbiter masses accounting for 39% and 21% respectively.

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# SPS Ballistic Launch Vehicle (Precursor) Mass Statement

SPS-1977

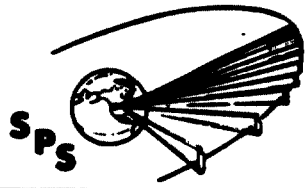
BOEING

|                        | <u>BOOSTER</u><br>(10 <sup>3</sup> KG) | <u>ORBITER</u><br>(10 <sup>3</sup> KG) |
|------------------------|----------------------------------------|----------------------------------------|
| STRUCTURE              | 60.36                                  | 49.80                                  |
| LANDING & AUX. SYSTEMS | 4.73                                   | 2.70                                   |
| ASCENT PROPULSION      | 50.69                                  | 17.93                                  |
| AUX. PROPULSION        | 1.12                                   | 2.60                                   |
| PRIME POWER            | 1.00                                   | .81                                    |
| HYD. CONV./DIST.       | .95                                    | .73                                    |
| ELECTRIC CONV./DIST.   | 1.36                                   | 1.00                                   |
| AVIONICS               | .88                                    | .97                                    |
| ECS                    | .94                                    | .94                                    |
| GROWTH                 | <u>8.70</u>                            | <u>6.48</u>                            |
| DRY MASS               | 130.73                                 | 63.95                                  |
| RESERVES & RESIDUALS   | <u>32.45</u>                           | <u>5.92</u>                            |
| LANDING MASS           | 163.18                                 | 69.87                                  |
| INFLIGHT LOSSES        | <u>22.70</u>                           | <u>18.16</u>                           |
| INERT MASS             | 185.88                                 | 108.02                                 |



**SPS BALLISTIC LAUNCH VEHICLE DDT&E COST**

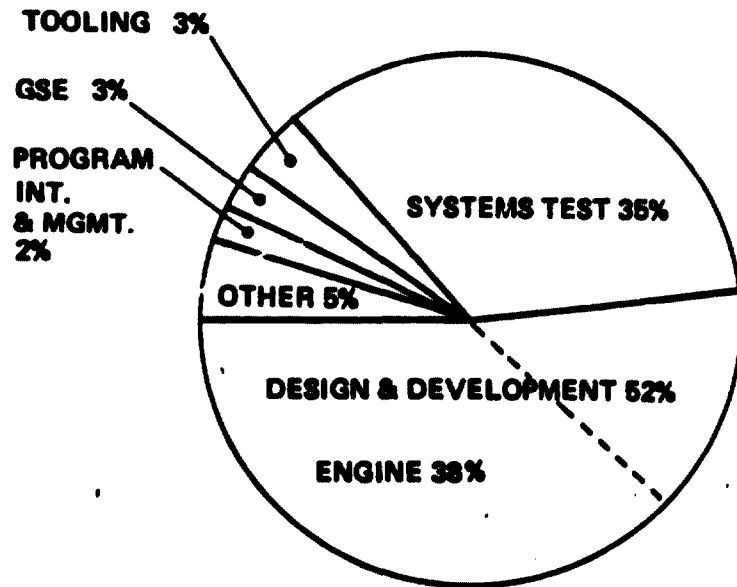
The DDT&E cost for flight hardware and associated ground support equipment for both stages is shown on the opposite chart. The total vehicle DDT&E cost is \$3.81 B. The booster DDT&E includes the cost of a new engine development. The costs of both stages reflect the use of some modified space shuttle subsystems.



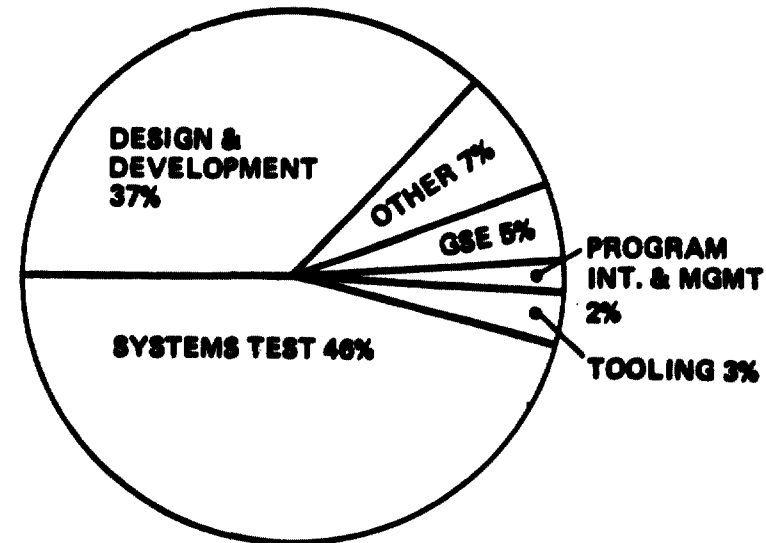
# SPS Ballistic Launch Vehicle (Precursor) DDT&E Cost

SPS-1970

ORRINO



BOOSTER DDT&amp;E = \$1,985M



ORBITER DDT&amp;E = \$1,828M

TOTAL VEHICLE DDT&amp;E = \$3.81B

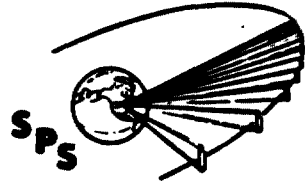
**SPS BALLISTIC LAUNCH VEHICLE PRODUCTION COST**

The theoretical first unit costs (TFU) of the booster and orbiter are shown on the adjacent chart. These were developed using the Boeing Parametric Cost Model (PCM). The booster and orbiter TFU costs are \$176.7M and \$202.7M respectively. The break-downs by subsystem are:

| SUBSYSTEM              | BOOSTER | ORBITER |
|------------------------|---------|---------|
| Structure              | 25%     | 30%     |
| Main Propulsion        | 34%     | 30%     |
| Avionics               | 9%      | 9%      |
| Landing & Aux. Systems | 4%      | 2%      |
| Other                  | 28%     | 29%     |

The estimated ground support equipment TFU costs are \$41.6M and \$44.8M for the booster and orbiter respectively.

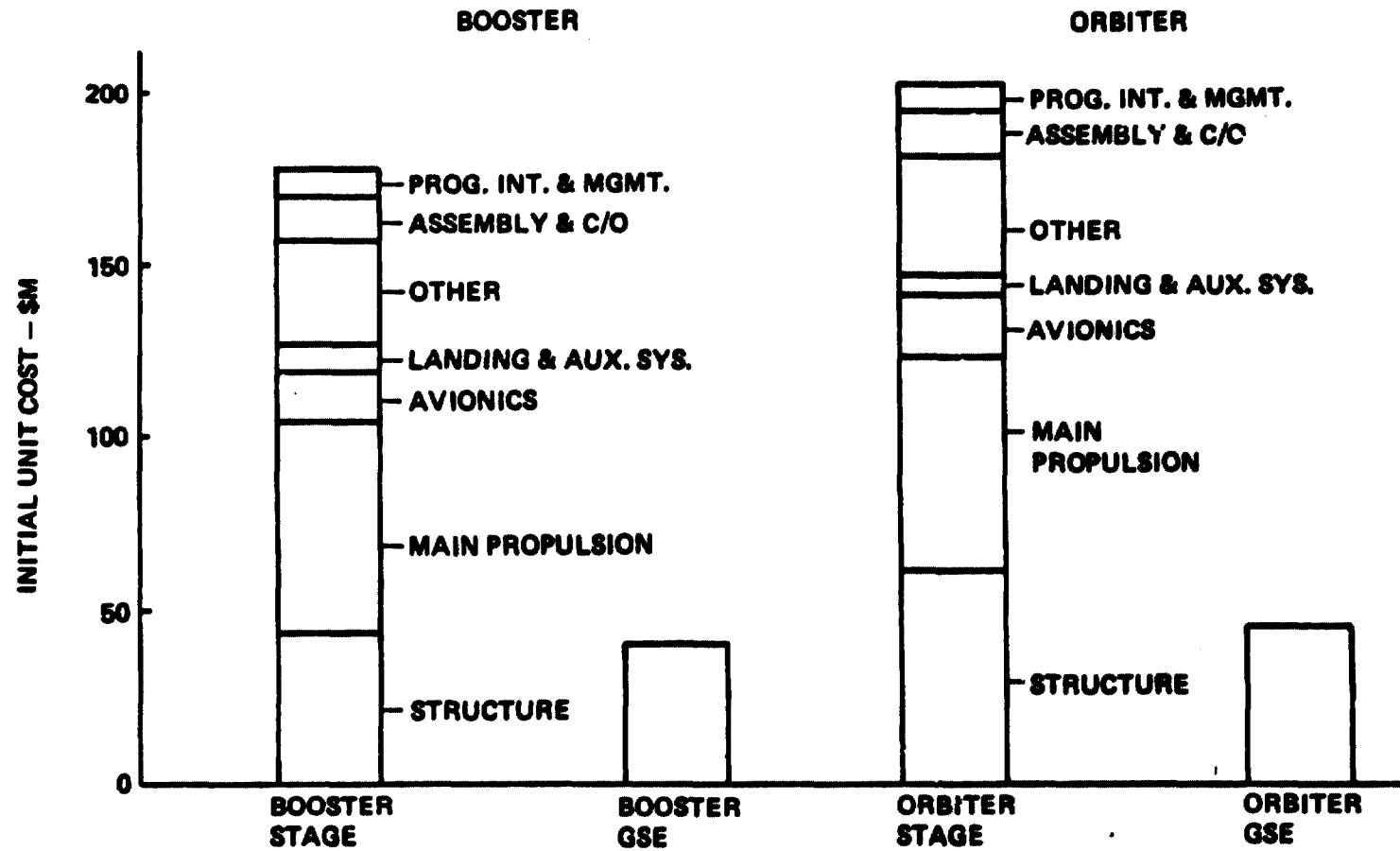
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SPS-1974

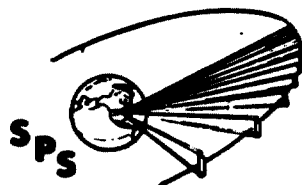
# SPS Ballistic Launch Vehicle (Precursor) Production Cost

BOEING



**SPS BALLISTIC LAUNCH VEHICLE  
AVERAGE COST/FLIGHT**

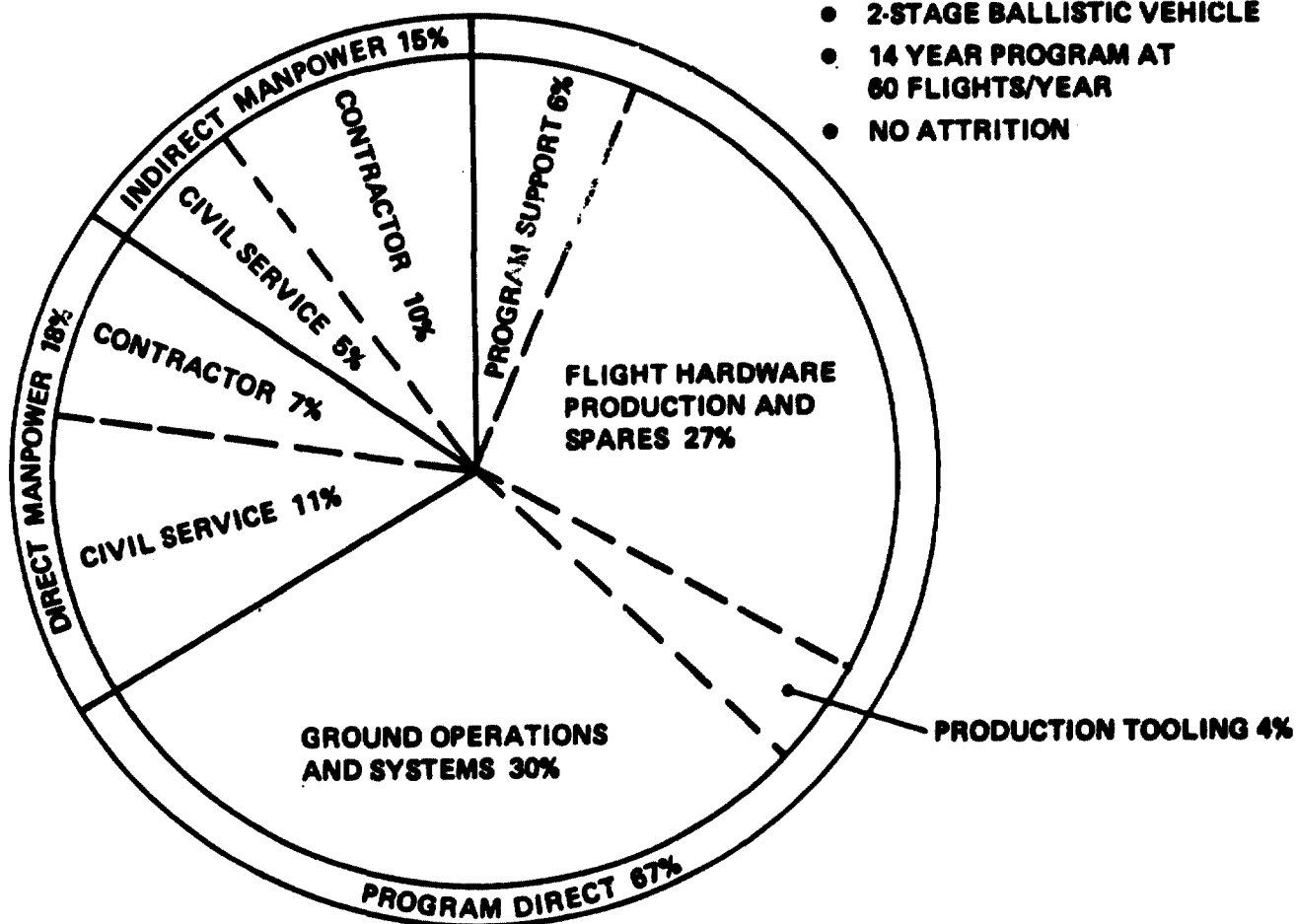
The cost/flight breakdown shown on the opposite chart is the average for a 14 year program at 60 flights per year. The major single elements are the flight hardware and ground operations costs accounting for 27% and 30% respectively.



SPS-1973

## Average Cost/Flight (Precursor 2-Stage Ballistic Recoverable)

BOEING



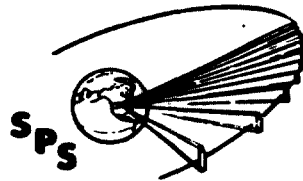
- 2-STAGE BALLISTIC VEHICLE
- 14 YEAR PROGRAM AT 60 FLIGHTS/YEAR
- NO ATTRITION

- AVERAGE COST/FLIGHT = \$9.986M

### **LEO TRANSPORTATION SUMMARY**

The highlights of the results from the LEO transportation activity are noted on the adjacent chart. The revised SPS two-stage winged vehicle concept has incorporated a number of desirable features such as booster flyback propulsion to enhance its operational characteristics. The economics of the winged vehicle appear attractive, and the additional benefit of a manned orbiter capability provides a single vehicle for both the cargo and crew transportation requirements.

The precursor 2-stage ballistic vehicle illustrates the results of one potential vehicle evolution path that begins with the SPACE SHUTTLE and progresses to a fully reusable vehicle in the 90,000 kg payload range. A key element in the development for any of these proposed 2-stage vehicles is the booster engine. Traditionally, a new engine development requires about eight years of development time compared to about five years for the airframe. As a result, a new booster engine will be the long lead time development item.



SPS-1978

D180-24071-3

## LEO Transportation Summary

**BOEING**

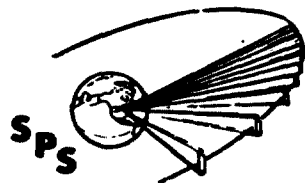
- **2-STAGE WINGED VEHICLE OFFERS:**
  - **AN AVERAGE COST/FLIGHT OF BETWEEN \$13.45M AND \$10.75M  
DEPENDENT ON SATELLITE INSTALLATION RATE (4/YEAR VS. 1/YEAR)**
  - **A 75 kg/m<sup>3</sup> PAYLOAD DENSITY**
  - **MINIMUM TURNAROUND TIME**
  - **MANNED CAPABILITY FOR CREW ROTATION/RESUPPLY**
- **PRECURSOR 2-STAGE BALLISTIC VEHICLE OFFERS**
  - **AN AVERAGE COST/FLIGHT FOR CARGO DELIVERY OF  
\$10M FOR 60 FLIGHTS PER YEAR**
  - **A MODEST DDT&E INVESTMENT OF \$3.8B**
  - **A BOOSTER STAGE FOR A SHUTTLE GROWTH CONCEPT  
(IMPROVED PERFORMANCE AND LOWER COST/FLIGHT)**
- **A NUMBER OF OPTIONS EXIST FOR LAUNCH VEHICLE EVOLUTION**
  - **LARGE THRUST BOOSTER ENGINE DEVELOPMENT PERIOD OF  
6-8 YEARS WILL BE THE LONG LEAD ITEM**



### MAINTENANCE AND OPERATIONS ANALYSIS

The topics to be covered in the maintenance and operations analysis are indicated. Initially, those items resulting in the greatest satellite power output loss will be identified. These items will be analyzed to determine the most desirable level of replacement at the satellite. The actual method of making the replacement will then be analyzed including the design impact on the antenna. Several aspects of the maintenance schedule will then be considered. The selected method will then be utilized in defining the maintenance mission characteristics. The selected approach in each of the above areas will then be incorporated into an overall maintenance system description followed by a summary of the maintenance operations identifying plant factor and annual maintenance cost per satellite.

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SPS-1904

D180-24071-3

# Maintenance and Operations Analysis

**BOEING**

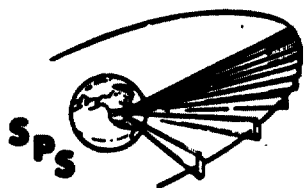
- ITEMS REQUIRING MAINTENANCE
- LEVEL OF REPLACEMENT
- REPLACEMENT CONCEPT
  - EQUIP REQ'D
  - ANTENNA DESIGN IMPACT
- MAINTENANCE SCHEDULE
  - HOW OFTEN
  - HOW RAPIDLY
  - WHEN
- MAINTENANCE MISSION
  - HABITAT LOCATION
  - REFURB LOCATION
  - TRANSPORTATION
- REFERENCE SYSTEM DESCRIPTION
- SATELLITE MAINTENANCE OPERATIONS SUMMARY

### **ITEMS REQUIRING MAINTENANCE**

A number of major components of the satellite have been analyzed for their nature of failures, mean time between failure, power loss per failure, and finally the power loss per year. The results of this analysis in terms of the power loss is presented.

As indicated, the component having the greatest impact in terms of power loss and in the time required to fix the failures is the klystron tube modules. DC/DC converters present a significant power loss although the number of failures is quite low and, consequently, require less repair time. The remainder of this analysis therefore will focus on the repair of the klystron tube modules.

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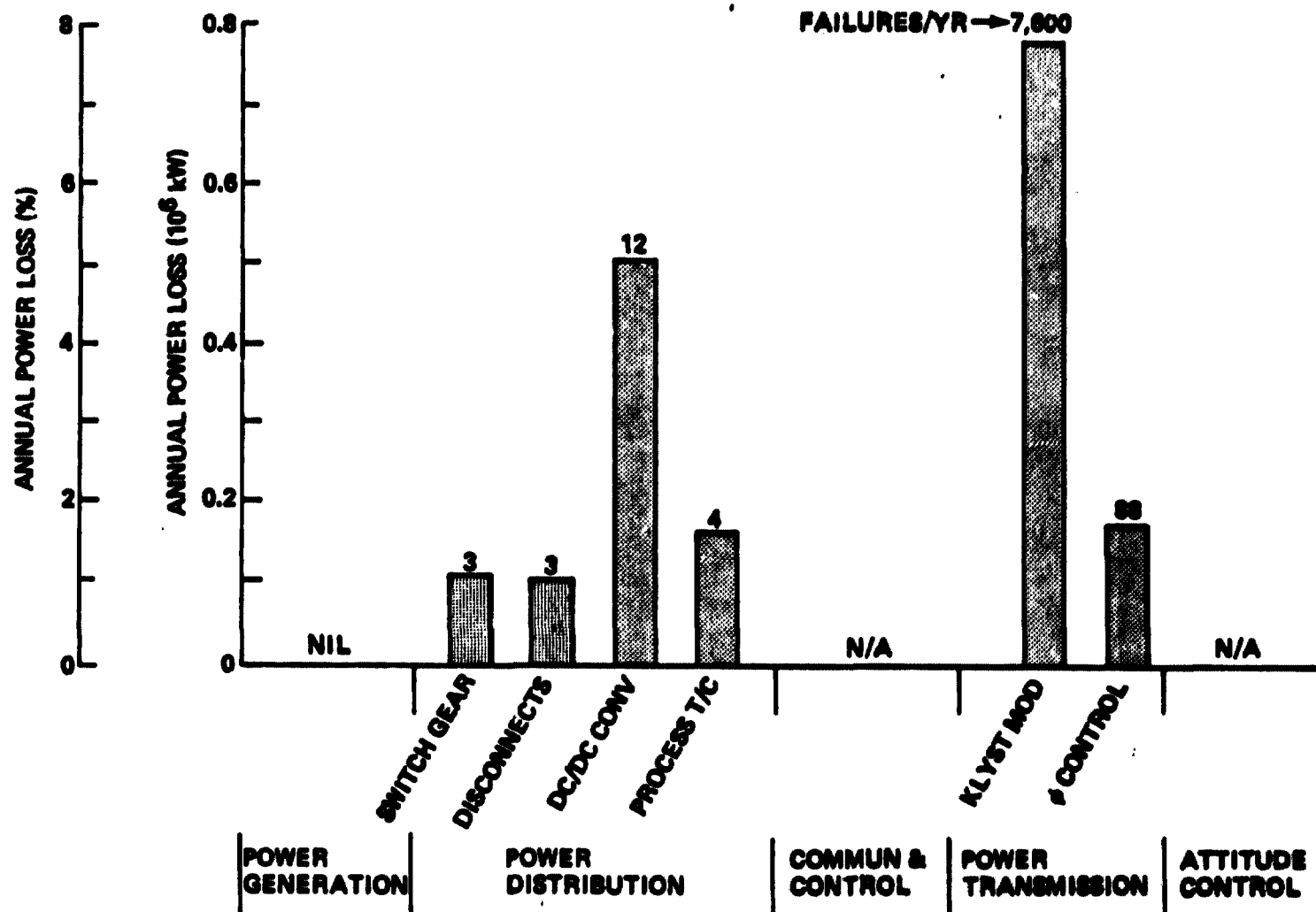


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## Items Requiring Maintenance

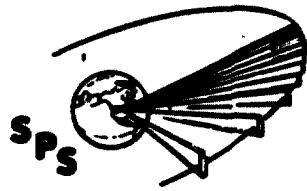
SPS-1935

BEING



**LEVEL OF REPLACEMENT OPTIONS AT SATELLITE**

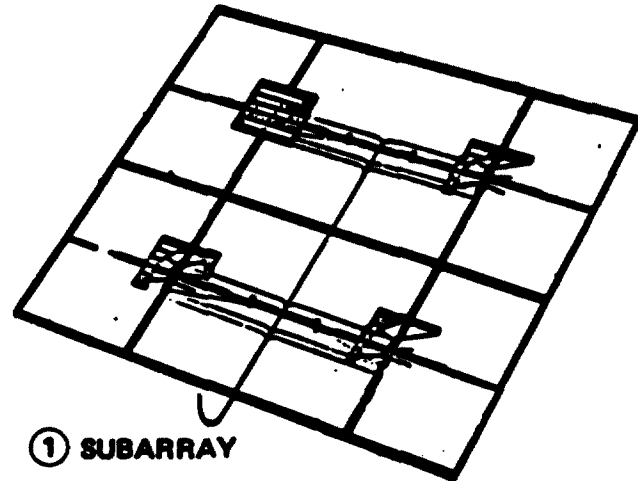
Several options exist for the replacement of a klystron tube module. The first of these options involves replacement of a complete subarray which may contain as many as 36 klystron tubes. The next level of replacement considered is that of a complete klystron tube module including the wave guide section associated with the module. Potential difficulty in segmenting the wave guide resulted in considering the third option which is the removal of the tube module and its thermal control system. The final option deals with removal of individual components which required the radiator to be sectionalized and have gimbaled panels. These options are compared in the following charts.



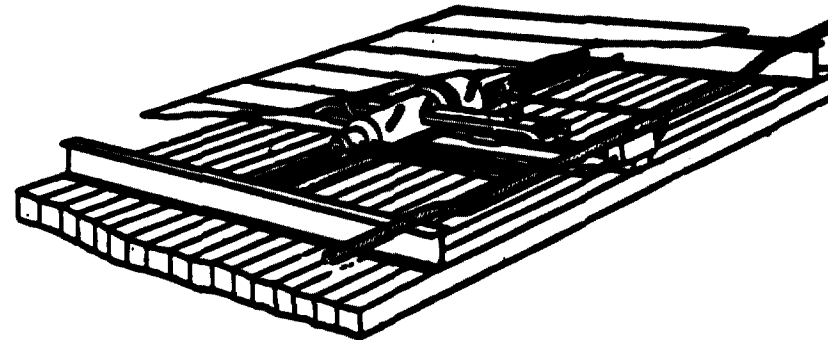
# Level of Replacement Options at Satellite

SPS-1911

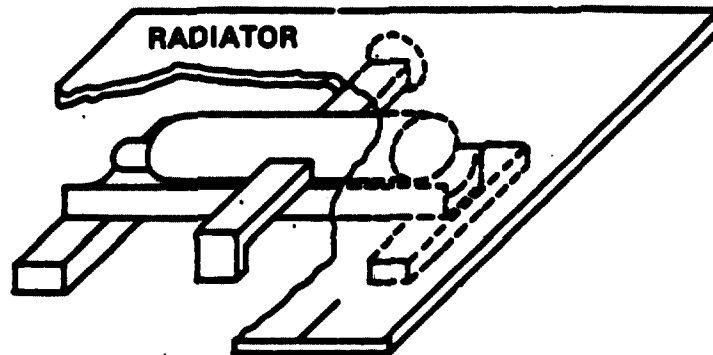
BOEING



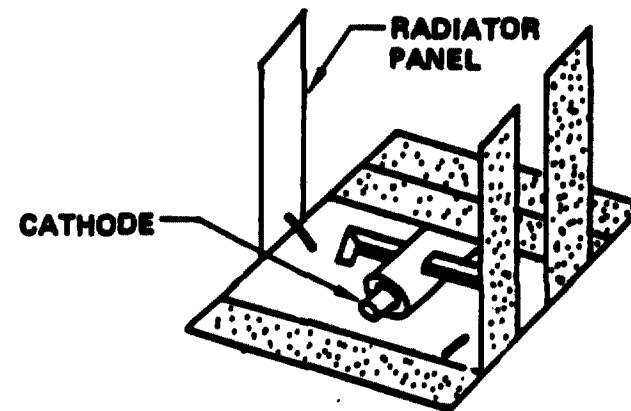
① SUBARRAY



② COMPLETE KLYSTRON MODULE INCLUDING WAVEGUIDE



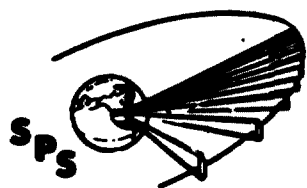
③ KLYSTRON TUBE PLUS THERMAL CONTROL



④ COMPONENTS

### **LEVEL OF REPLACEMENT SELECTION**

The level of replacement selected is that of the klystron tube module plus its thermal control system. The rationale for selection of this option is indicated. Actual removal of the tube module involves access through holes in the radiator to reach the distribution wave guide attachment bracket which secures the module to the distribution wave guide. Once this attachment is released the module is free to be removed.

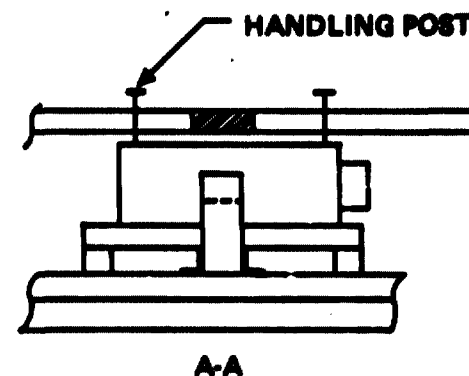
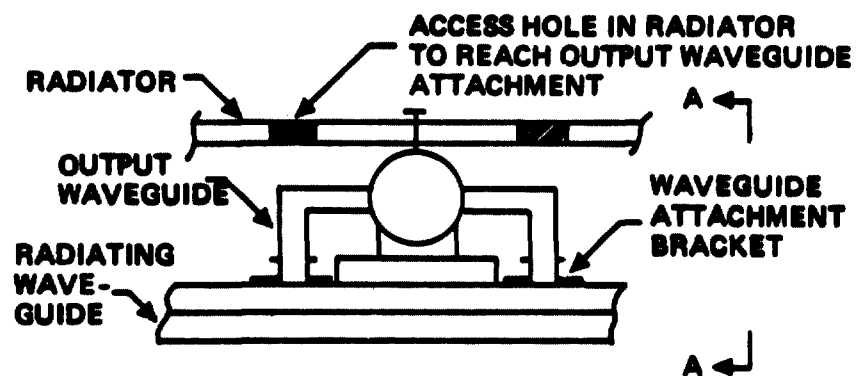


# Level of Replacement Selection

SPS-1912

BOEING

## ● SELECTION: TUBE PLUS RADIATOR



## ● RATIONALE

- SUBARRAY – REQUIRES REMOVAL OF 30% (2400) OF UNITS FOR 4% (7600) OF TUBES  
– MAINTENANCE TIME AND MASS PENALTY
- COMPLETE MODULE – MASS PENALTY AND INCREASED ALIGNMENT DIFFICULTY
- COMPONENTS – RADIATOR DESIGN COMPLEXITY, TEST VERIFICATION PROBLEM

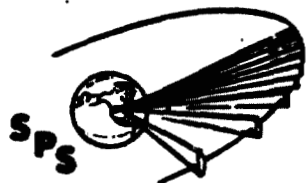


### **TUBE MODULE REPLACEMENT OPTIONS**

Using the klystron tube module as the level of replacement, it is now possible to analyze various options relative to the method of replacing the module. In all cases, it is assumed the satellite is shut down while maintenance is performed.

Two options involve servicing the antenna from the back side (non-radiating). One of these has the repair vehicle moving horizontally through the secondary structure until the failed tube is reached. Another option has the repair vehicle moving horizontally through the primary structure and then vertically through the secondary structure to reach the failed tube. The last option has the servicing done from the front side through use of moveable overhead platform.

Each of these options are discussed in more detail in subsequent charts.



D180-24071-3

# Tube Module Replacement

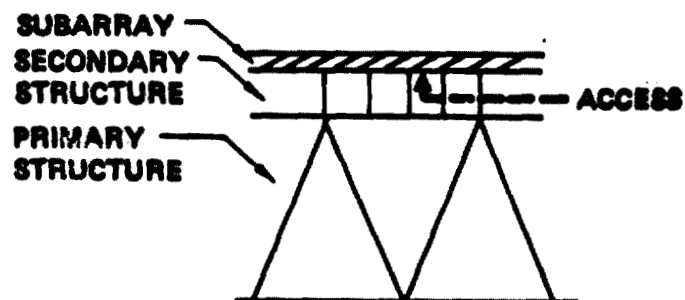
SPS-1919

BOEING

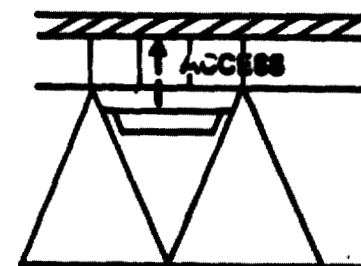
## ● OPTIONS

### ● BACKSIDE SERVICING

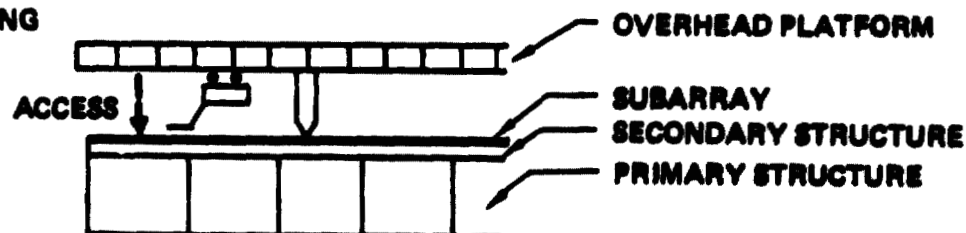
#### ● HORIZONTAL ACCESS THROUGH SECONDARY STRUCTURE



#### ● VERTICAL ACCESS THROUGH SECONDARY STRUCTURE



### ● FRONTSIDE SERVICING

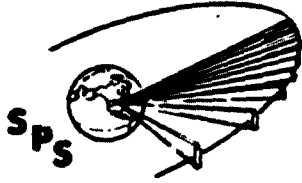


## ● KEY CONSIDERATIONS

- ACCESS
- EQUIPMENT REQUIRED
- ANTENNA DESIGN IMPACT

**HORIZONTAL ACCESS FOR TUBE MAINTENANCE**

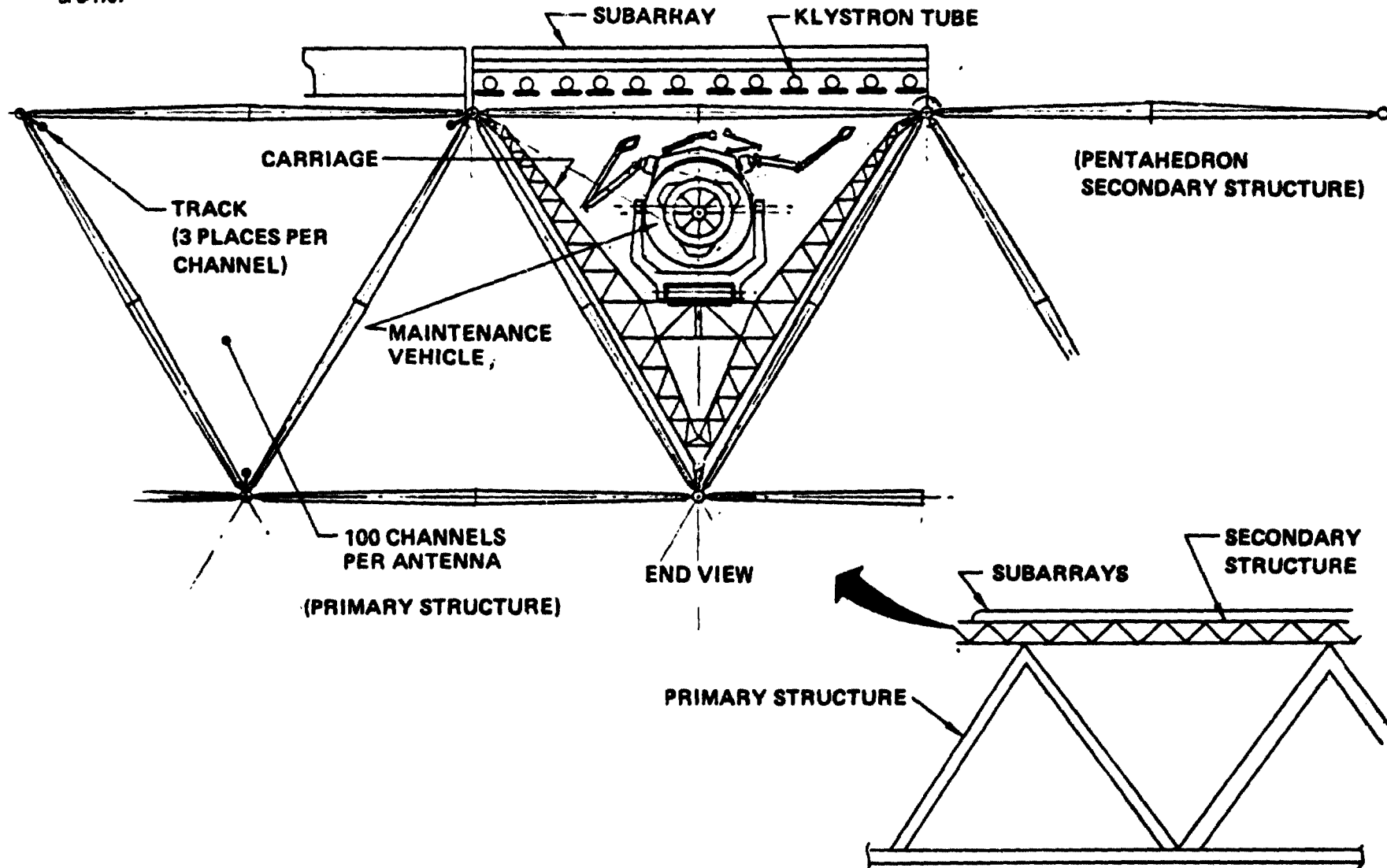
The next three charts deal with reaching the failed klystron tube module by horizontal travel through the secondary structure. In this concept a two-man maintenance vehicle is mounted to a carriage that in turn is supported from three tracks within a channel formed by the secondary structure. Across the antenna a total of 100 channels exist, each requiring a built-in track system. Early estimates indicate as many as 10 maintenance vehicles to be required within the 100 channels.



# Horizontal Access For Tube Maintenance

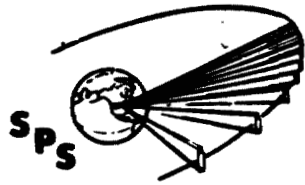
SPS-1787

BOEING



**HORIZONTAL ACCESS FOR TUBE MAINTENANCE – SIDE VIEW**

A side view of the maintenance vehicle in the horizontal access concept is shown. A gimbal system between the carriage and the maintenance vehicle allows positioning of the maintenance vehicle relative to the tube above. The platform also provides an area for storing klystron tube modules.

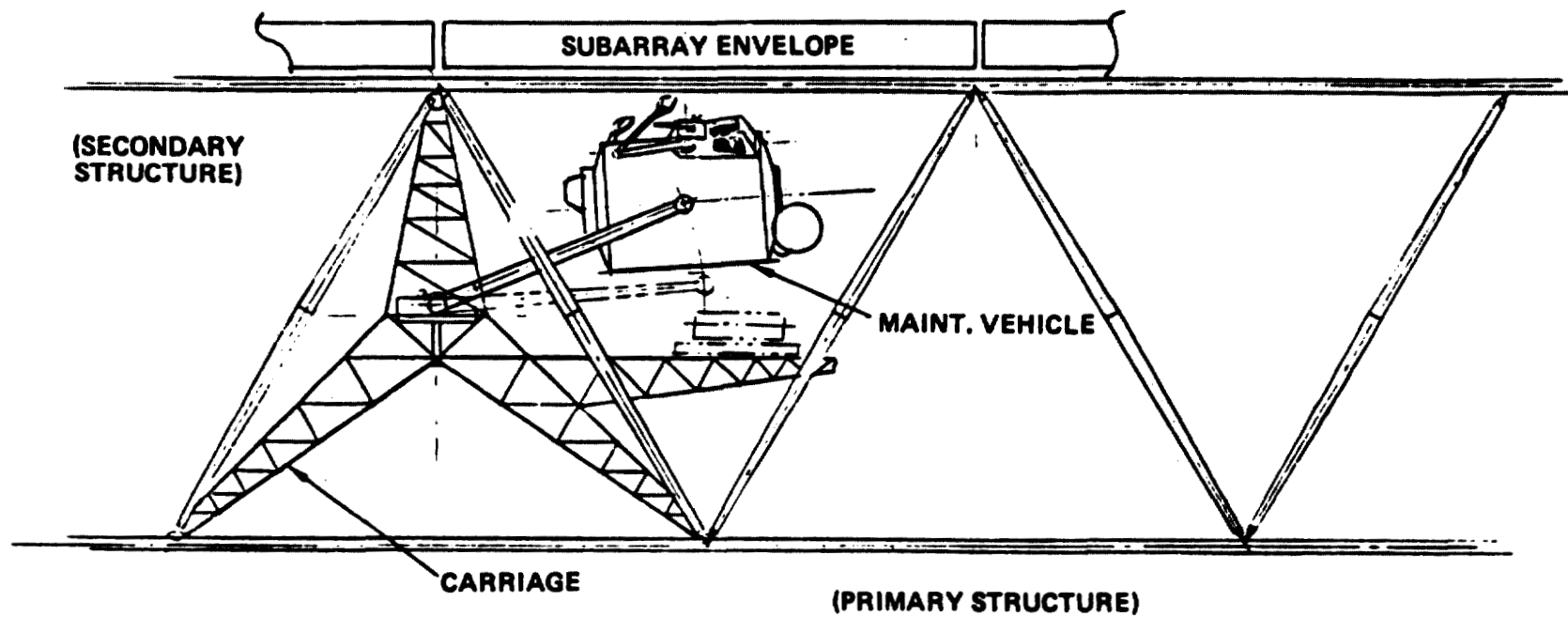


D180-24071-3

## Horizontal Access For Tube Maintenance

SPS-1788

BOEING

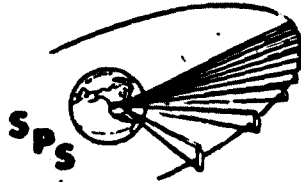


SIDE VIEW

### **HORIZONTAL ACCESS – SECONDARY STRUCTURE OPTIONS**

Several secondary structure options were analyzed for the horizontal access option. The tetrahedron structure was used throughout part 2. Part 3 of the study considered a pentahedron type structure. Both structures are deployable. Comparison of the structures show that the pentahedron only has one diagonal between the maintenance vehicle and the klystron tube. The tetrahedron structure would have 2 diagonals, potentially resulting in a longer repair time due to diagonal removal and replacement. Other factors favoring the pentahedron structure was that a little more space is available for the maintenance equipment within the channel and the track installation would be somewhat easier.

# Horizontal Access Secondary Structure Options

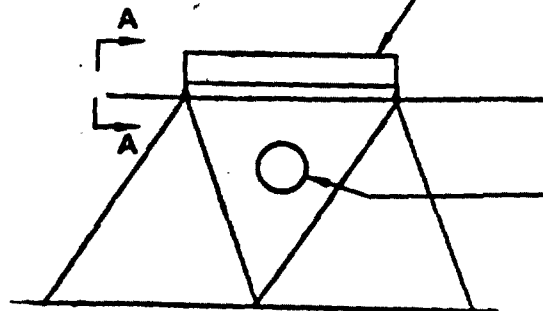


SPS-178

BOEING

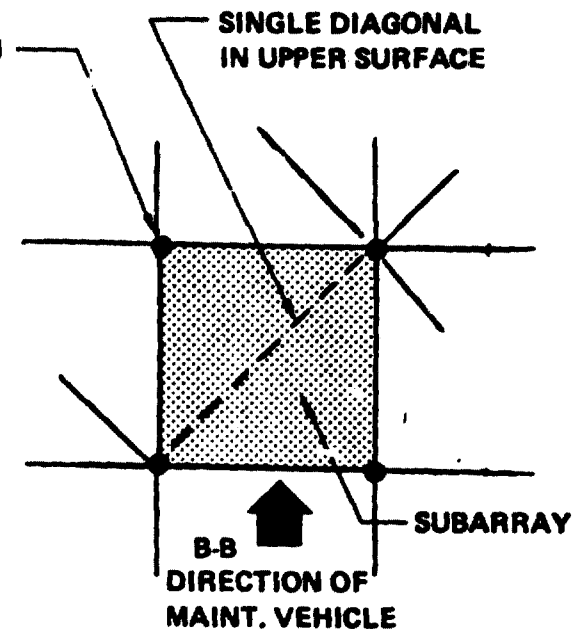
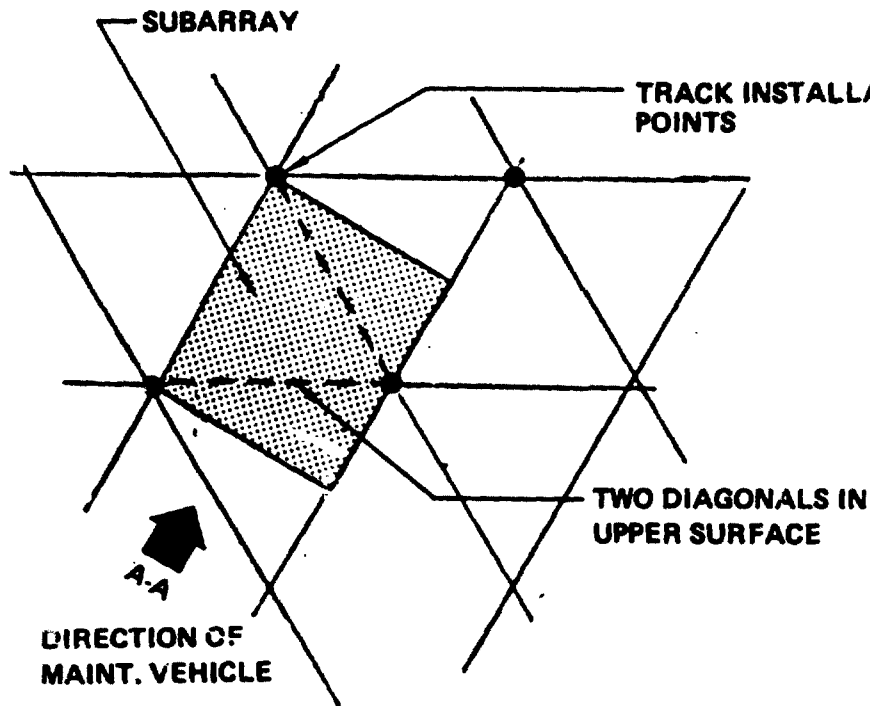
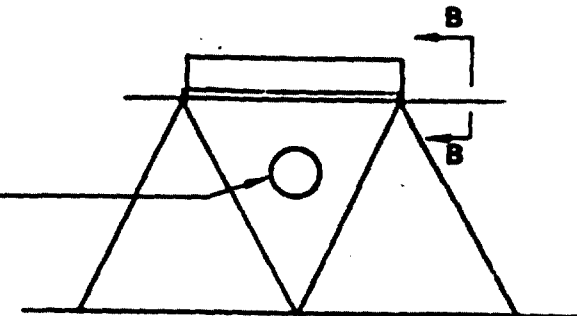
**TETRAHEDRON**

SUBARRAY



REPAIR VEHICLE

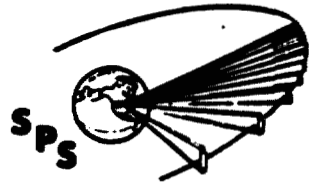
**PENTAHEDRON**





**VERTICAL ACCESS FOR TUBE MAINTENANCE**

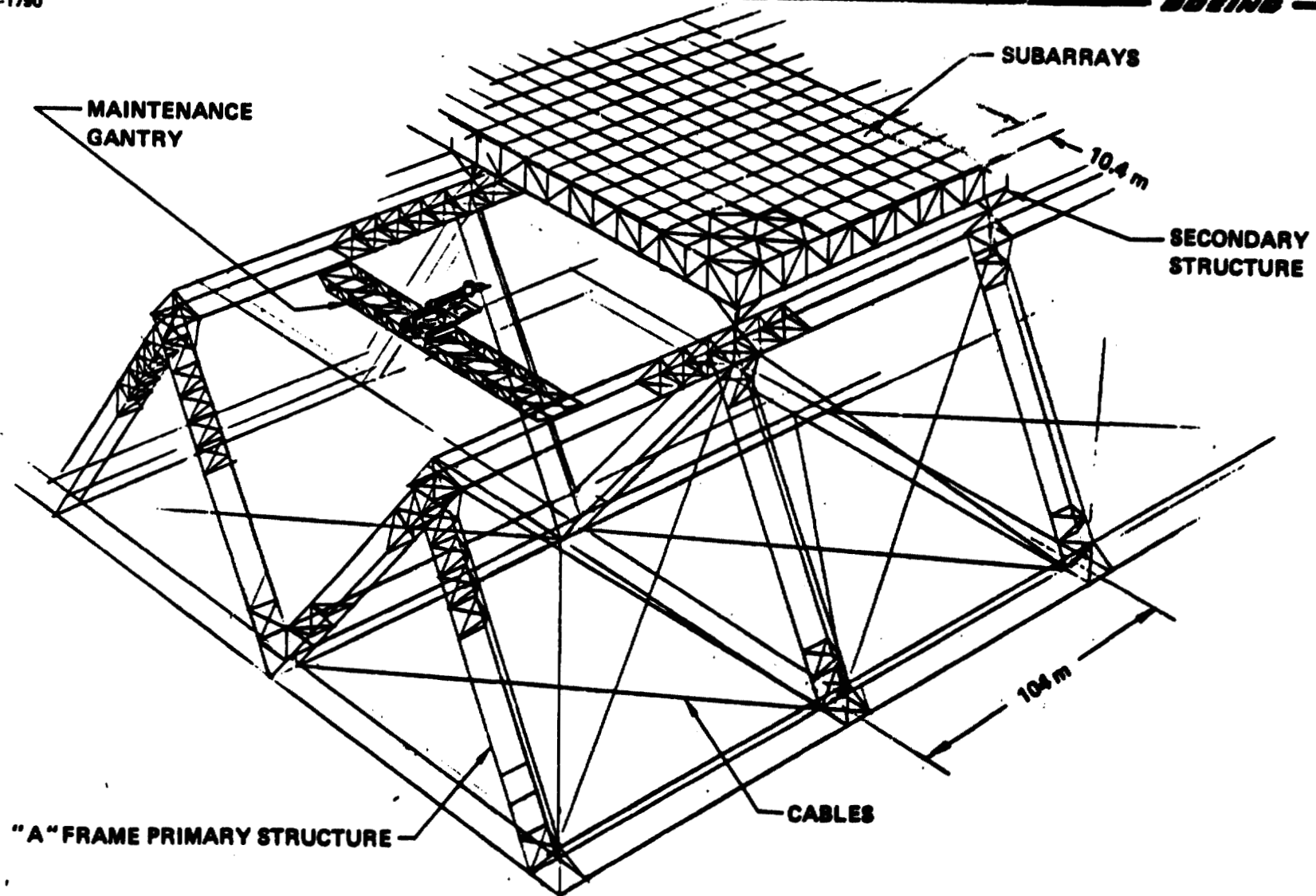
For the vertical access approach, the antenna has a total of 10 channels in which maintenance gantries can be mounted. Attached to each of the gantries is the maintenance vehicle which reaches up through the secondary structure to reach the failed klystron tubes. In this option, a cubic (hexahedron) structure has been found to offer the greatest clearance for the maintenance vehicle.



SPS-1790

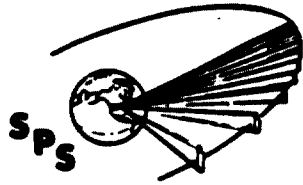
## Vertical Access For Tube Maintenance

BOEING



**VERTICAL ACCESS FOR TUBE MAINTENANCE**

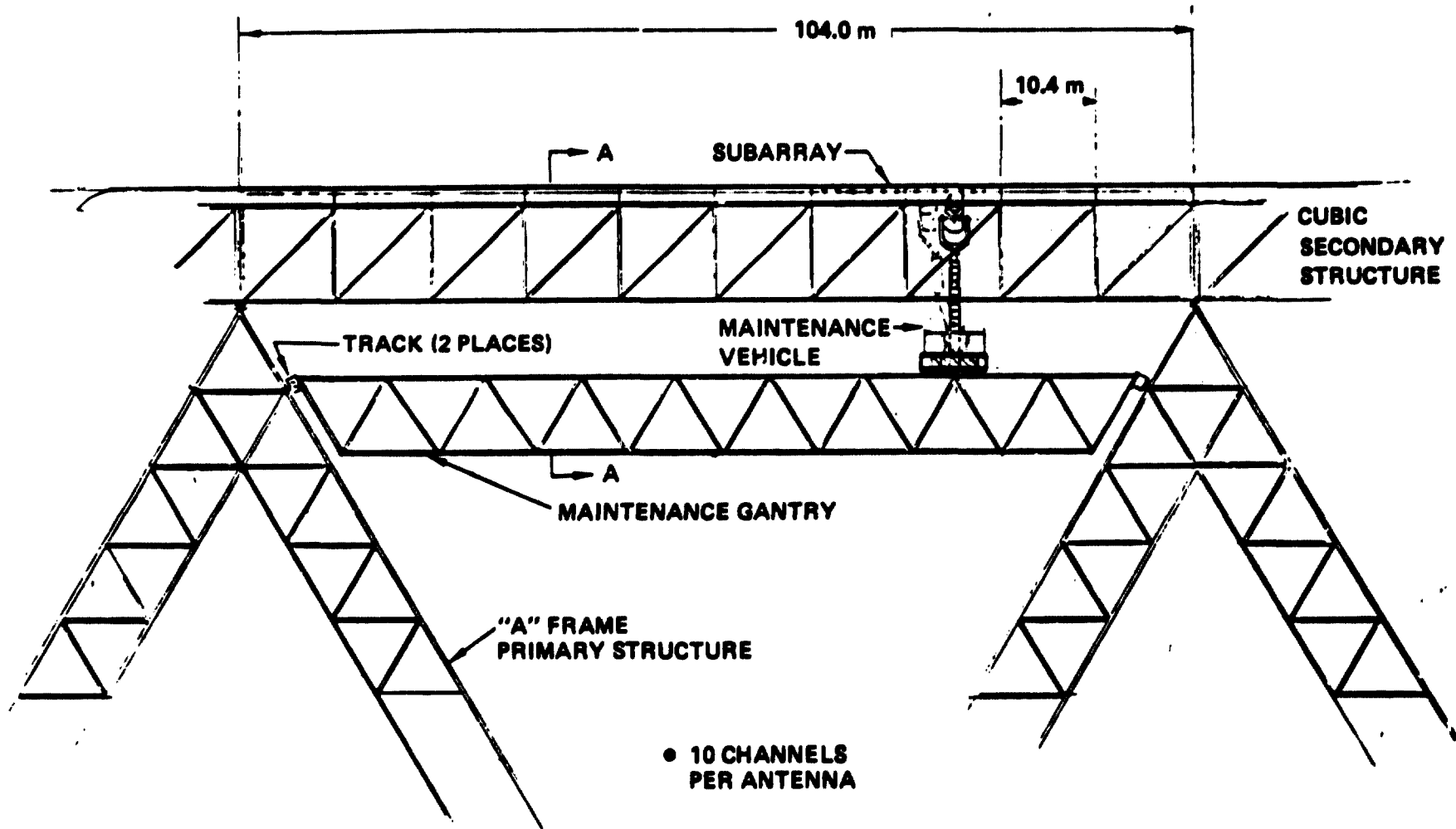
The next three charts deal with the back side servicing concept involving vertical access for klystron tube maintenance. The overall concept is illustrated in this chart. The primary structure is a frame design forming ridges that allows free unobstructed movement of the maintenance gantry moving horizontally across the antenna. Stability of the primary structure in one direction is provided by the cubic secondary structure above the primary structure while stability in the opposite direction is provided through guy-wire cabling.



# Vertical Access For Tube Maintenance

SPS-1791

BOEING

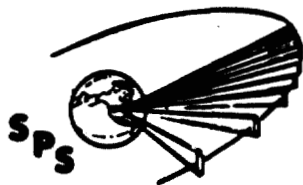


### **VERTICAL ACCESS MAINTENANCE VEHICLE**

Additional detail of the cubic secondary structure and the maintenance vehicle is presented with the maintenance gantry shown moving along in the direction of the channel. The gantry itself will be designed to transport all of the spare klystron tubes necessary for a given shift. The maintenance vehicle consists of a hinged boom and a 2-man crew cabin with manipulators. A small klystron rack is also attached to the boom to eliminate the need for the manipulators to reach back down to the gantry for a new tube. In the case of a 36 tube subarray, as many as 3 tubes may require replacement.

The cubic secondary structure is deployable and to satisfy packaging constraints has its vertical and diagonal members telescoping approximately 25%, while horizontal members have knee joints. A packaging density comparable to the part 2 tetrahedron structure is anticipated and is estimated at 70 kilograms a cubic meter.

Using this concept, a tube replacement time of 45 min. is expected, which includes removal and replacement of two diagonals (one in lower and upper surface of secondary structure), removal and replacement of one klystron tube module, and movement to the next failed klystron tube estimated at a distance of 2 subarrays away or 20 meters.

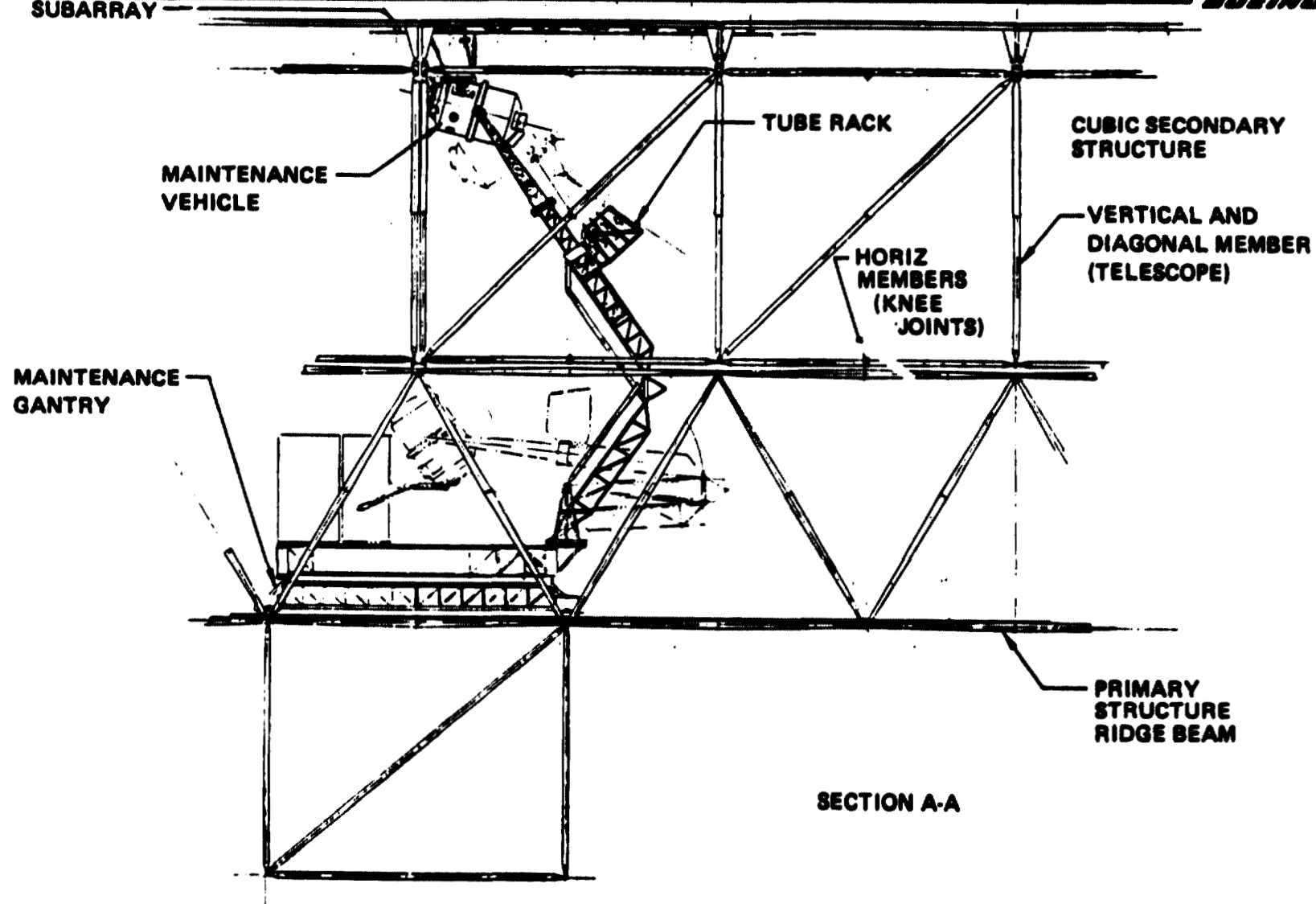


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# Vertical Access Maintenance Vehicle

SPS-1792 SUBARRAY

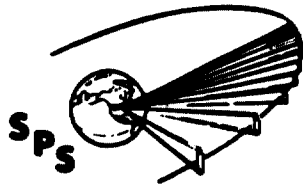
BOEING



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**ANTENNA FRONT SIDE MAINTENANCE CONCEPT**

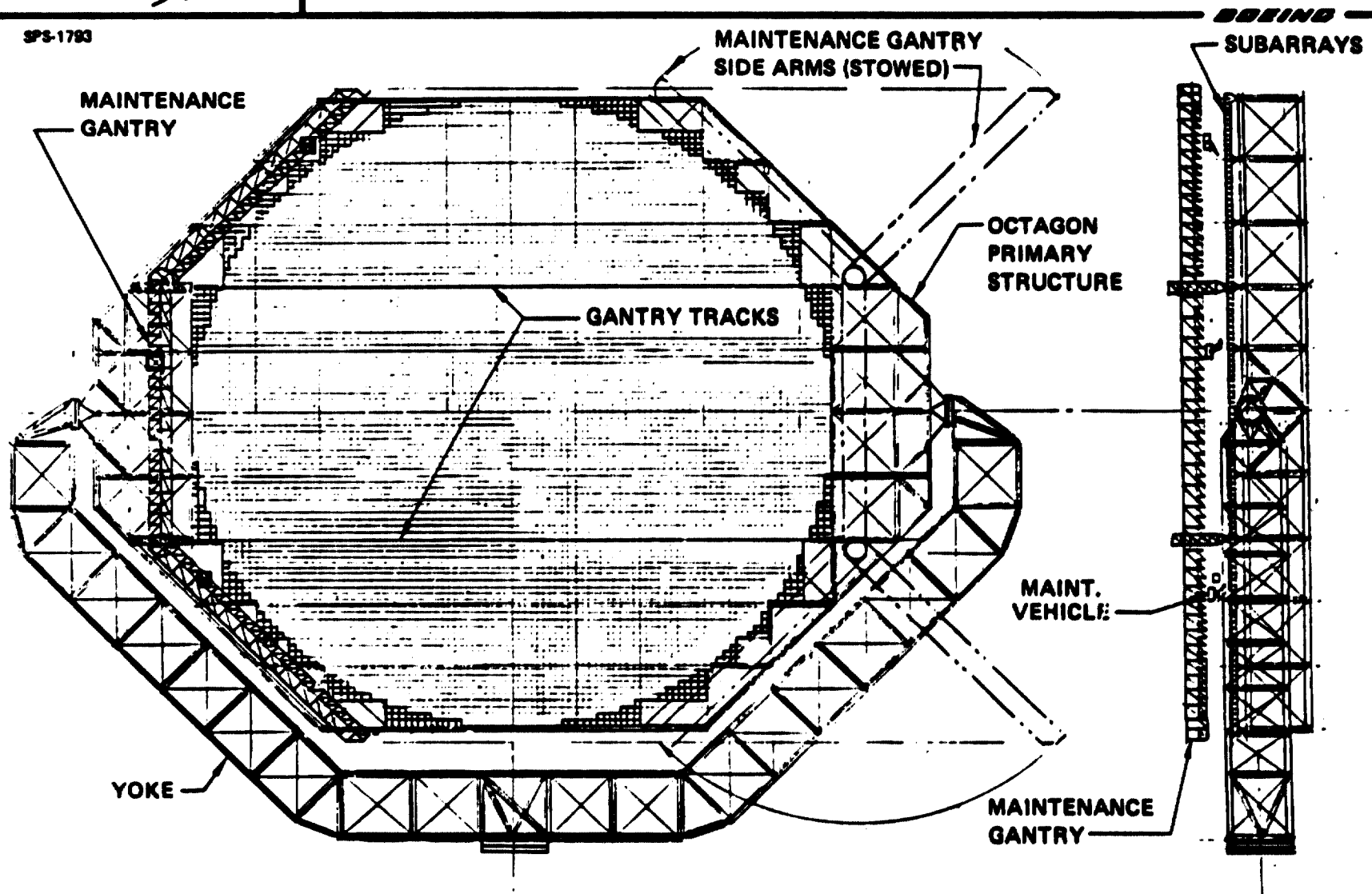
*The next two charts deal with the front side servicing concept. This concept consists of an overhead maintenance gantry that is supported from two tracks running across the antenna. Supported from the platforms are the actual repair vehicles. The antenna design for this option has been altered to form an octagon shape. The actual primary structure for the antenna is oversized to allow the maintenance gantry to be moved out of the way when the antenna is in the operating condition.*



D180-24071-3

## Antenna Frontside Maintenance Concept

SPS-1793

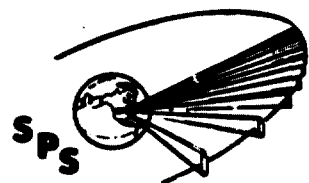






### **MAINTENANCE VEHICLE CONCEPT—FRONT SIDE ACCESS**

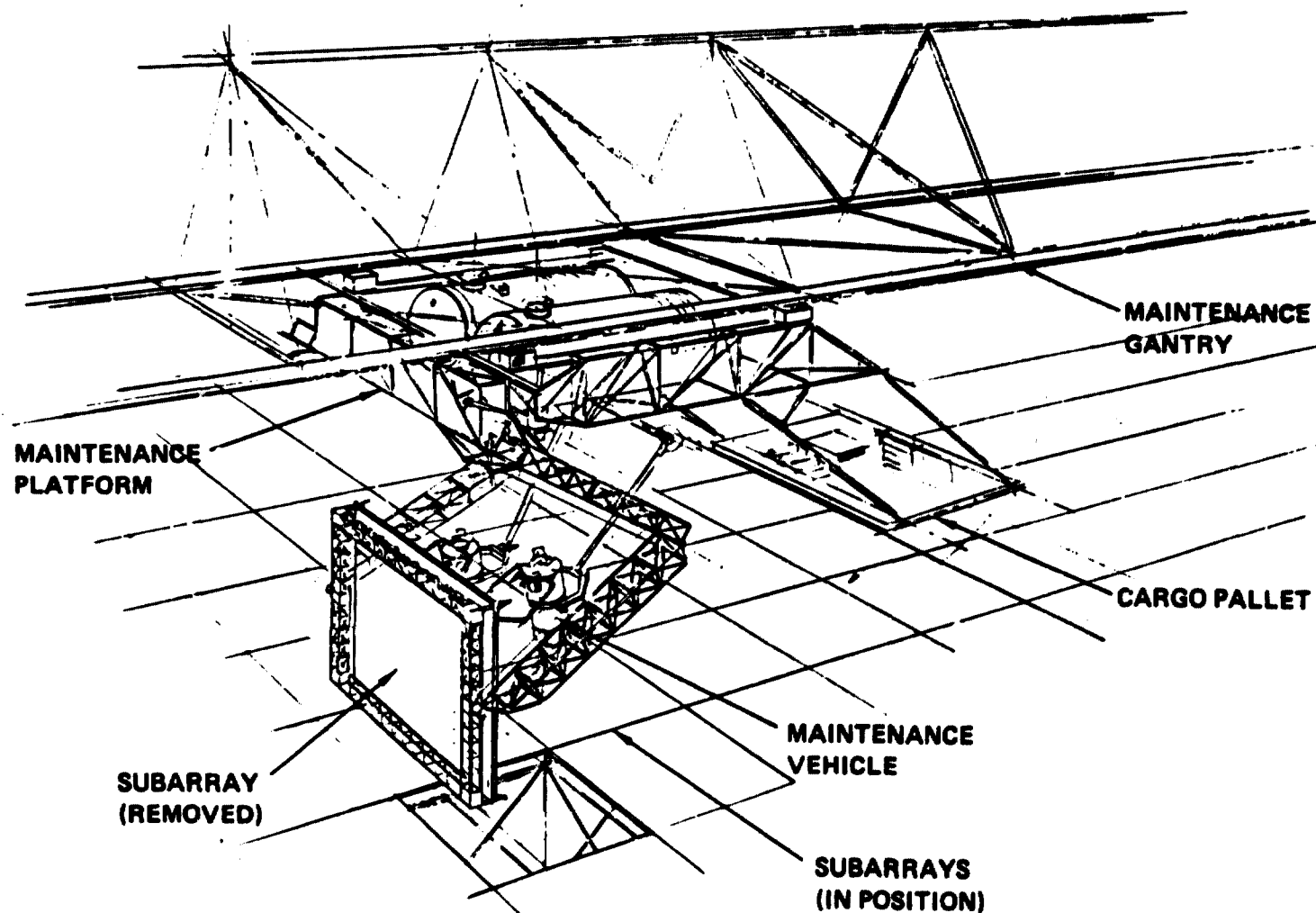
The actual replacement of the klystron tubes in this option requires the subarrays to be removed as shown. Once removed, the maintenance vehicles can then remove the failed klystron tubes. When all failed tubes of a given subarray are repaired the subarray is replaced.



SPS-1788

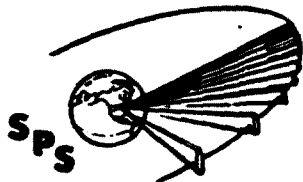
# Maintenance Vehicle Concept Frontside Access

**BOEING**



### **REPLACEMENT CONCEPT SELECTION**

The selected klystron tube module replacement concept uses vertical access through the cubic secondary structure which is attached to the "A"-frame primary structure. The advantages of this option over the other options are indicated. The chief advantage over the horizontal access option was that only 10 track systems were required while less repair time and less mass were the chief advantages over the front side repair concept.



SPS-1913

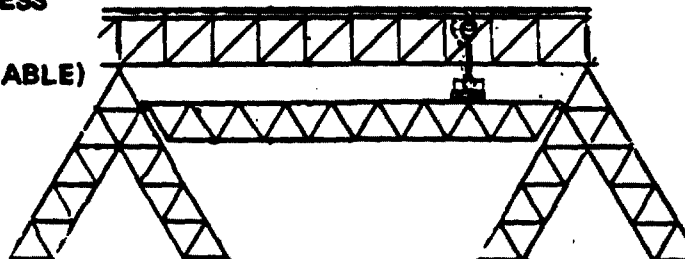
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## Replacement Concept Selection

BOEING

### ● SELECTION: VERTICAL ACCESS

- CUBIC SECONDARY
- STRUCTURE (DEPLOYABLE)
- "A" FRAME PRIMARY STRUCT



### ● RATIONALE: ADVANTAGES OVER

#### ● HORIZONTAL ACCESS

- 10 TRACK SYSTEMS VERSUS 100
- EASE IN INSTALLING TRACK SYSTEM
- ABILITY TO POSITION REPAIR VEHICLE DIRECTLY UNDER ALL TUBES
- EASIER REMOVAL OF COMPLETE SUBARRAY IF NECESSARY

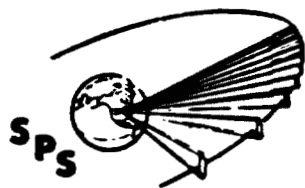
#### ● FRONTSIDE ACCESS

- LESS DOWNTIME (LOST REVENUE)
- REPAIR VEHICLES ARE DECOUPLED
- SMALLER ANTENNA PRIMARY STRUCTURE AND YOKE
- NO ANTENNA PERFORMANCE PENALTY

**MAINTENANCE SCHEDULE—HOW OFTEN TO REPAIR**

The maintenance schedule employed for servicing the antenna deals with (1) how often to repair, (2) how rapidly to repair once the antenna is reached, and (3) when (time of day and year) the antenna should be repaired.

In this chart the effect of how often illustrates the average power output factor and the associated lost revenue. As the antenna is repaired more frequently, the average power output is higher and accordingly the lost revenue is less. It should be noted, however, that for frequencies more often than 6 months the benefit gets proportionately less. Further maintenance analysis will utilize the 6 month repair frequency concept.



SPS-1903

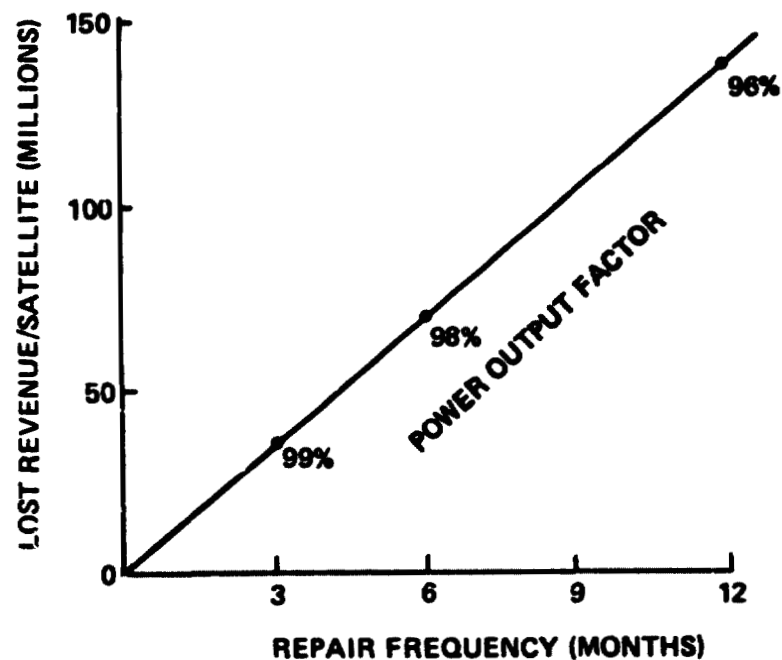
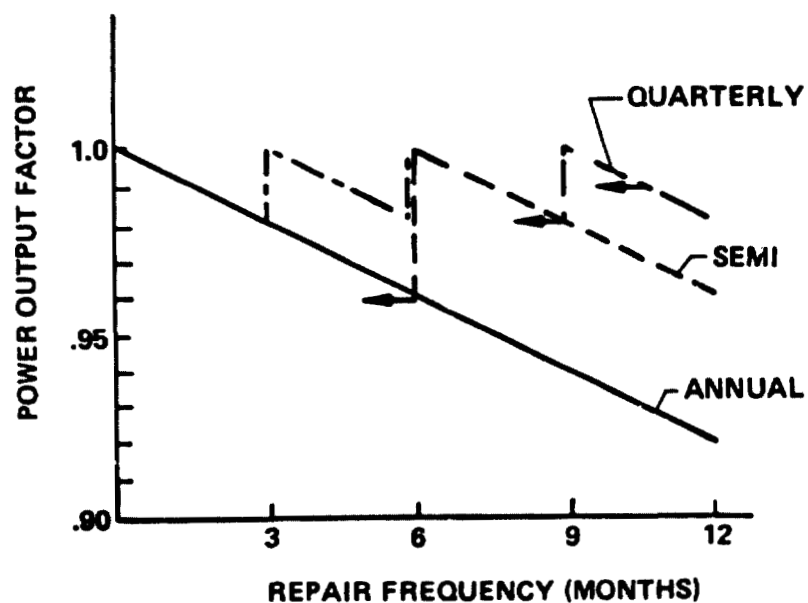
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# Maintenance Schedule

## How Often To Repair

BOEING

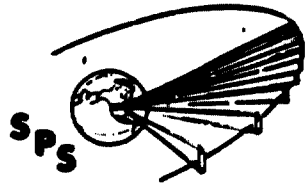
- KEY PARAMETER: POWER OUTPUT FACTOR
- 1% TUBE FAILURE GIVES 2% POWER LOSS
- 4% TUBE FAILURE PER YEAR
- ANNUAL REVENUE ~ \$3.5 BILLION @ 100% OUTPUT FACTOR



**MAINTENANCE SCHEDULE—HOW RAPIDLY TO REPAIR**

Once the satellite is reached, a key issue then becomes how quickly the repair should be performed. The key factor in this issue then is how many repair crews and associated equipment are provided. The factors to be considered in this assessment are indicated and are discussed in the subsequent charts.

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# Maintenance Schedule How Rapidly To Repair

**BOEING**

**KEY PARAMETER: CREW SIZE**

**FACTORS:**

- DOWNTIME—LOST REVENUE
- CREW LOGISTICS COST
- MAINTENANCE PROVISIONS COST
  - HABITATS
  - REPAIR EQUIP
- PRACTICALITY



**DOWN TIME COST**

Down time cost deals with the actual time the satellite is shut down and not transmitting power. Using the semi-annual maintenance approach a total 3800 tubes per satellite involving 2850 hours of work are involved. Characteristics for each repair crew are indicated and result in a down time of 14 days per semi-annual repair when using one repair crew per satellite. Larger numbers of crews reduce the down time and the resulting down time cost.

# Downtime Cost

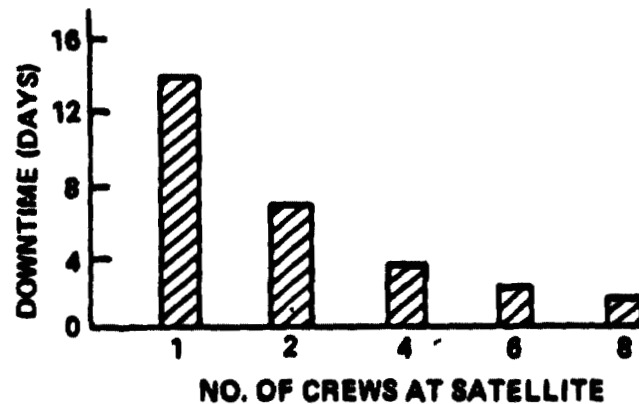


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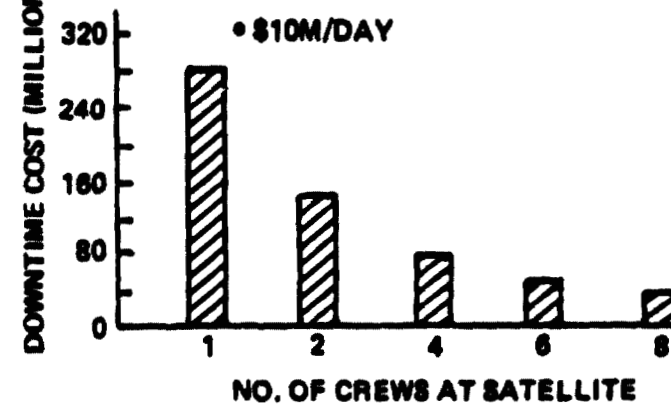
BEING

- SINGLE TUBE REPLACEMENT TIME: 45 MIN
- TOTAL TUBE REPLACEMENT TIME: 2850 HRS
  - SEMI-ANNUAL REPAIR
  - 3800 TUBES
- REPAIR SYSTEMS AND OPERATIONS PER CREW
  - 10 REPAIR VEHICLES/ANTENNA (FIXED EQUIP)
  - 2 SHIFTS (TOTAL OF 60 PEOPLE)
  - 10 HOURS/SHIFT
- RESULTS PER SATELLITE

DOWNTIME DURATION

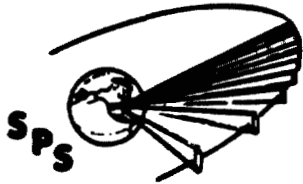


ANNUAL DOWNTIME COST (\$10<sup>6</sup>)



**LOGISTICS COST**

**Logistics cost, meaning the transportation of crew and their supplies, goes up with the number of crews as indicated. It must be noted however that although the higher number of crews means more cost, that complement of people also has a greater potential in terms of how many satellites can be repaired in a given increment of time. Performing an equal amount of work (repair of satellites) therefore is necessary in order to have a legitimate comparison.**

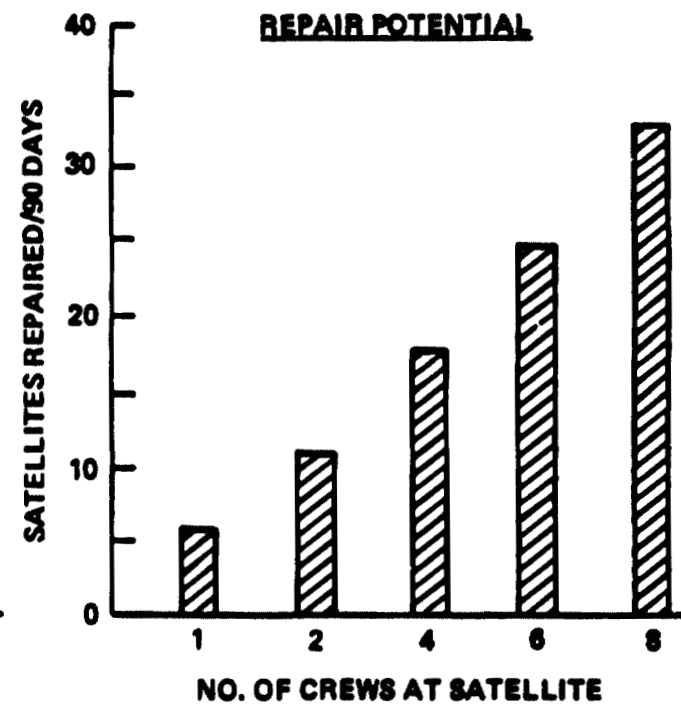
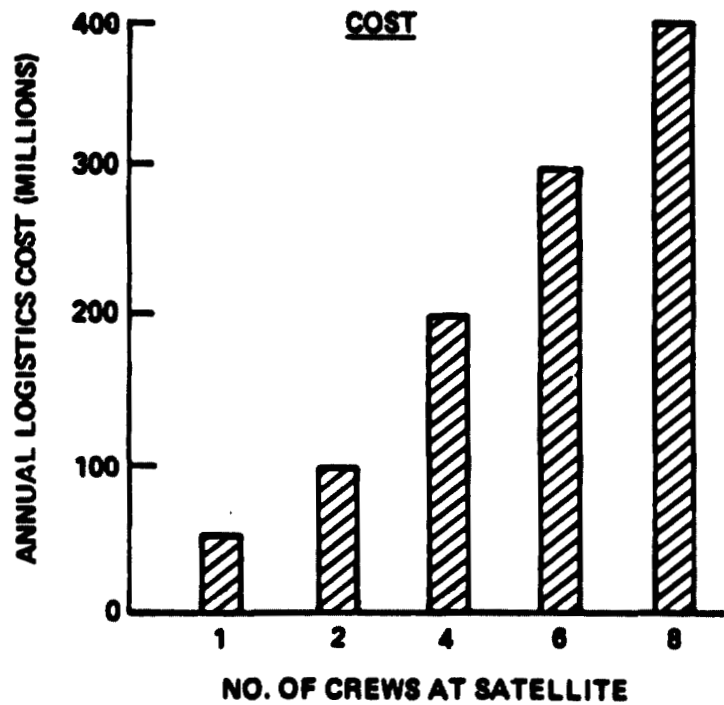


SPS-1907

# Logistics Cost

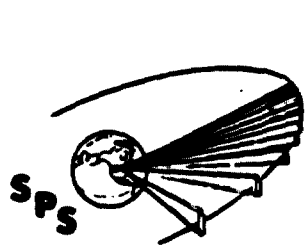
BOEING

- ONE SATELLITE IN ORBIT
- A CREW IS 60 PEOPLE
- 2 REPAIR SORTIES/SATELLITE/YEAR
- SHUTTLE GROWTH TO LEO
- CHEM OTV TO GEO



### **MAINTENANCE PROVISIONS**

The third key factor in the issue of repair speed is the amount of maintenance provisions. The two key provisions in this case are the movable crew habitats and the actual repair equipment (maintenance gantry and vehicle) that is fixed to the satellite. Major characteristics of each of these systems are provided including mass and cost estimates. A 15% capital charge factor (same as used in costing the satellite) is used for the write-off.



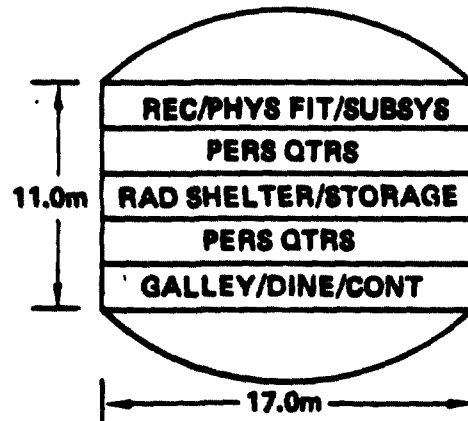
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## Maintenance Provisions

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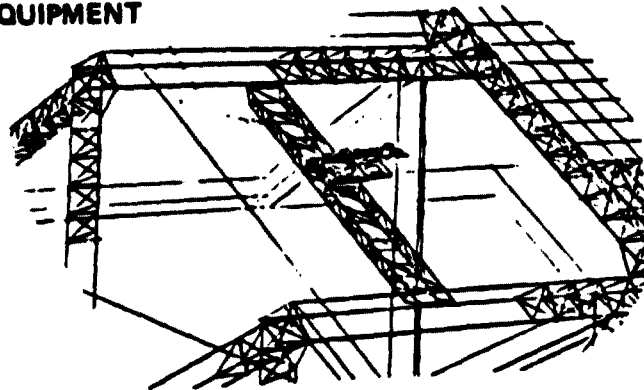
**BOEING**

### • CREW HABITAT



- ONE HABITAT PER 60 PEOPLE
- MODIFIED CREW QUARTERS MODULE
- 240,000 kg
- \$240 MILLION INVESTMENT
- 15% CAPITAL CHARGE

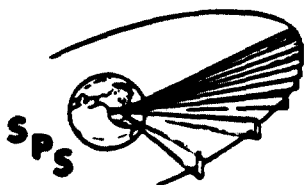
### • REPAIR EQUIPMENT



- GANTRY/REPAIR VEHICLE (EA)
  - 5,000 kg
  - \$20 MILLION TFU
  - 15% CAPITAL CHARGE
- TRANSPORTATION COST  
EARTH TO GEO SELF POWER = \$45/kg

**CREW SIZE COST**

Combining the factors of down time cost, logistics cost, and maintenance provision cost allows the selection of the crew size in terms of annual cost. For the case of only one satellite in orbit, two crews per satellite gives the least cost. For the 10 satellites in orbit case, two and four crews give about the same annual cost. It may be noted that in the 10 satellite case the down time and the permanently installed repair equipment at the satellite have a much greater impact on total cost. From this trend it would appear that with a greater number of satellites in orbit, a larger number of crews may be more cost optimum. Selection of the number of crews at the satellite, however, requires a final consideration in terms of the practicality discussed on the next chart.

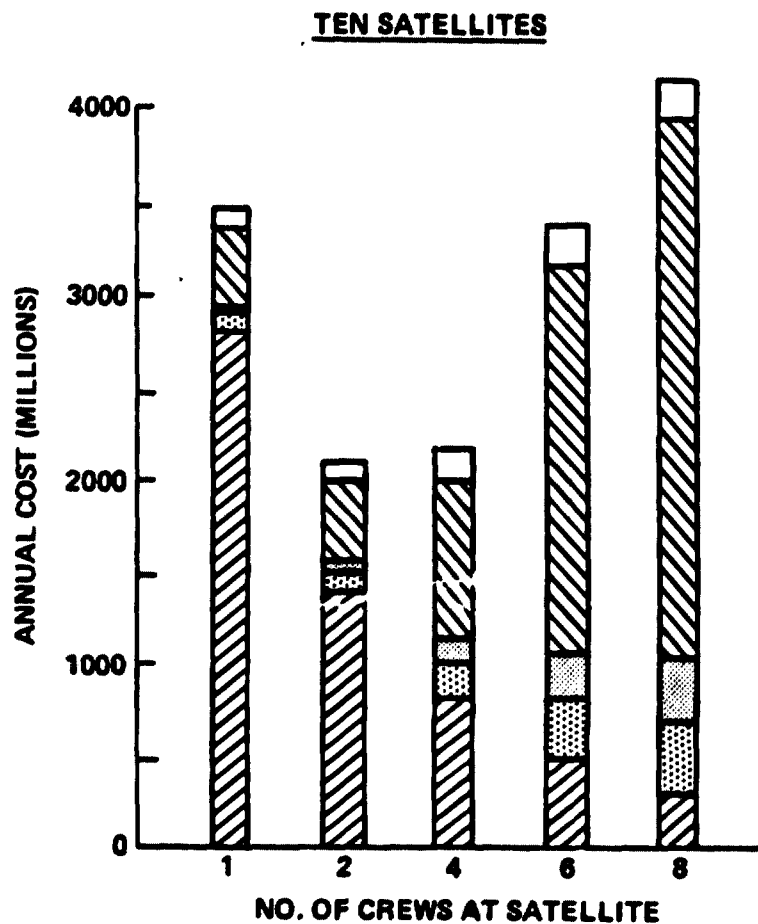
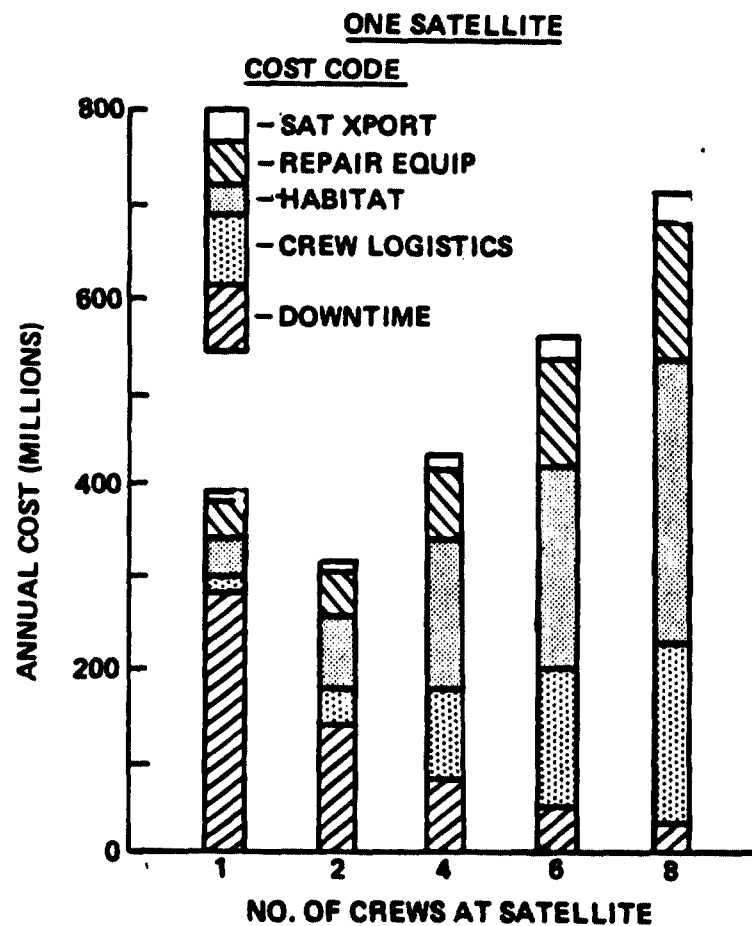


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# Crew Size Cost

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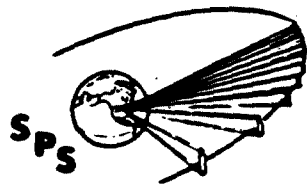


**CREW/EQUIPMENT UTILIZATION—PRACTICALITY**

This chart attempts to illustrate the relative complexity of actually integrating into an antenna various numbers of repair equipment. It may be remembered that each crew utilizes 10 repair vehicles, 20 hours a day (two shifts). For the case of one crew, 10 repair vehicles are installed in each antenna during the antenna construction at the LEO construction base. Once the crew completes the repair of antenna number 1 they move to antenna number 2 for its repair.

Use of two crews at the satellite allows one crew to work on each antenna in parallel. Consideration of four crews requires additional repair vehicles to be installed as indicated. When 6 or more crews are considered several repair vehicles must be put on each gantry which results in potential dynamics problems and/or multiple gantries installed per channel.

From the standpoint of cost, four crews appear reasonable, and from a crew equipment utilization viewpoint the same number of crews may be the practical limit. Four crews per satellite are consequently selected for the remainder of the maintenance analysis.

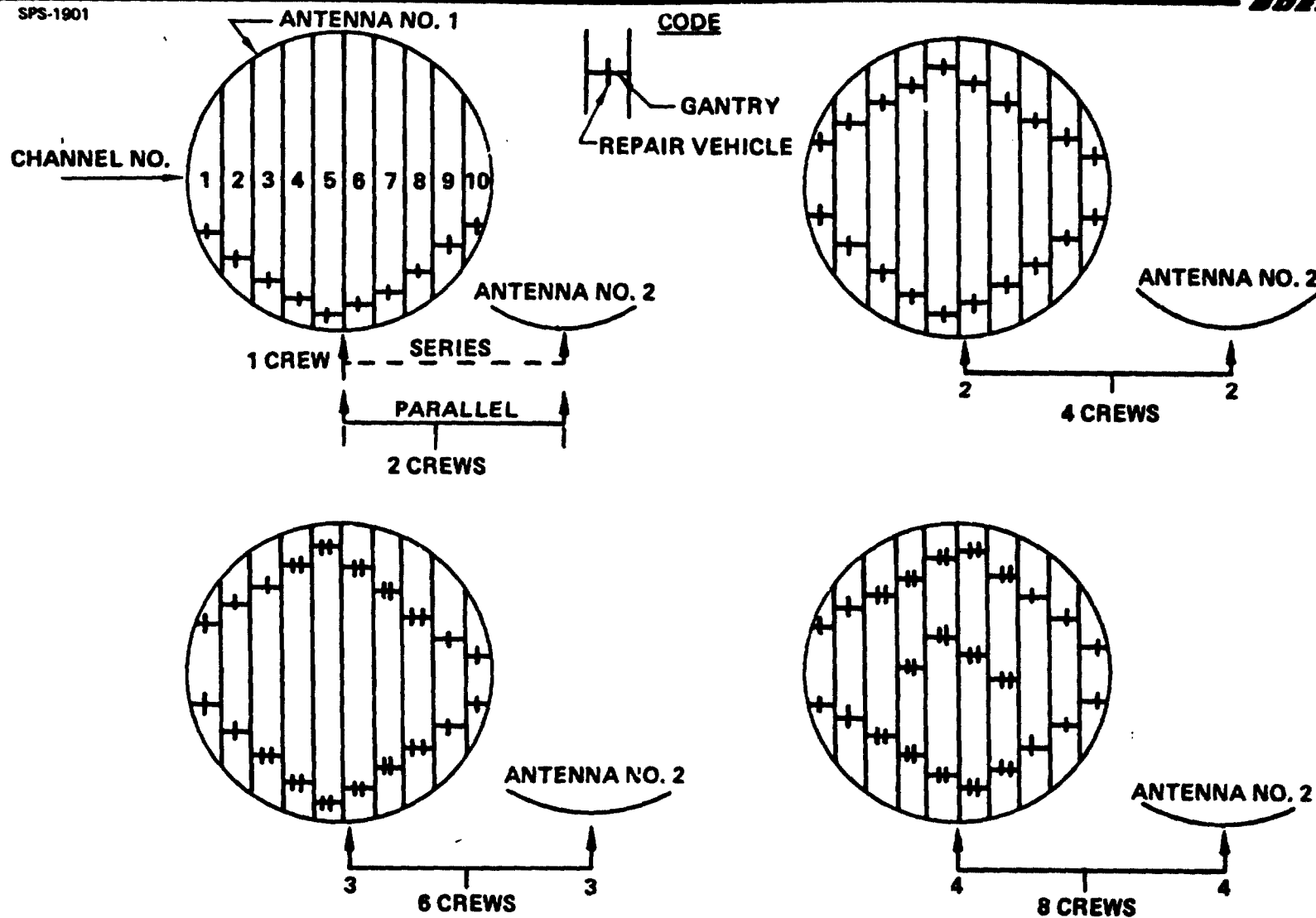


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D180-24071 3

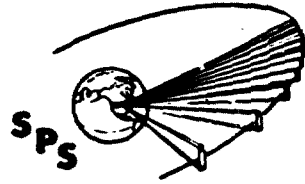
# Crew/Equipment Utilization Practicality

BOEING



**MAINTENANCE SCHEDULE—WHEN TO REPAIR**

A third key issue regarding the maintenance schedule is when the repair is to be performed. Two subissues are involved in this primary issue. The first of these deals with the time of day maintenance is to be performed, while the other concerns itself with the time of year. Both will be discussed in subsequent charts. Key considerations in these issues are similar to past issues with the additional consideration of the impact on the ground power grid.



SPS-1914

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## Maintenance Schedule When to Repair

BOEING

- **KEY ISSUES:**

- TIME OF DAY—DOES IT PAY TO LIMIT REPAIR TO A FEW HOURS AROUND THE DAILY OCCULTATION DURING EQUINOX?
- TIME OF YEAR—DOES IT PAY TO REPAIR JUST DURING THE EQUINOXES?

- **KEY CONSIDERATIONS:**

- DOWNTIME COST
- LOGISTICS COST
- GROUND POWER GRID IMPACT

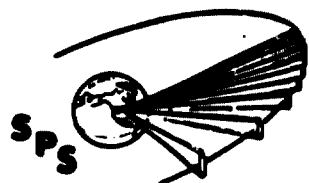
- **KEY VARIABLE:**

- NUMBER OF SATELLITES

**WHEN TO REPAIR--ORBIT GEOMETRY**

A key factor in analyzing the "when to repair" issue is the orbit geometry involved since down time is a major factor. The time of day issue deals with utilizing the daily occultation time occurring during the equinox when an average of 50 minutes per day of mandatory satellite down time occurs.

The time of year issue deals with confining the repair to the equinox which lasts 44 days. In the extreme case, one can also consider the case of doing the repair work beginning at the beginning of one equinox and lasting until the beginning of the second equinox which means 180 days available to repair the satellites.

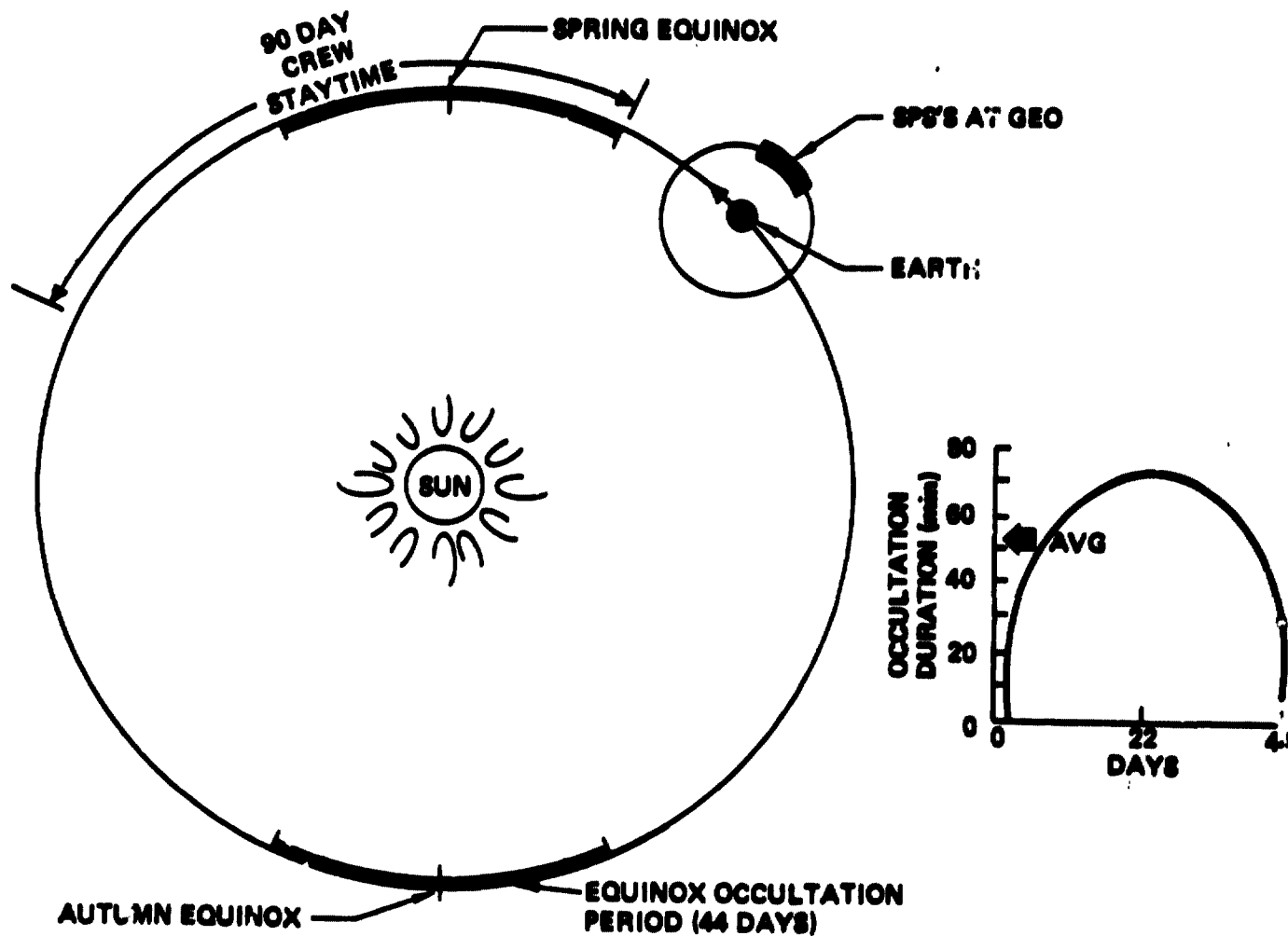


SPS-1918

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# When to Repair Orbit Geometry

SPS-1918

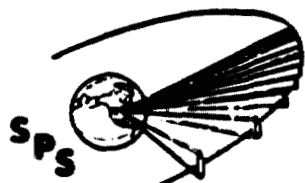


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### **DAILY REPAIR SCHEDULE OPTIONS**

Two primary options were considered relative to the daily repair schedule. The first of these called "Continuous" involves shutting down the satellite and then working on it until finished using four crews per satellite. Using four crews per satellite results in a repair time of  $3\frac{1}{2}$  days. A transportation time of  $\frac{1}{2}$  day to the next satellite and  $\frac{1}{2}$  day to activate the repair equipment at the satellite results in this one group of four crews repairing 10 satellites during an equinox period.

An alternate daily repair schedule would be an intermittent approach which centers the repair around the 50 minute occultation period per day. In the case investigated, about 4.5 hours are added on either side of the occultation period making it a 10 hour work day as has been used throughout the study. Since only one 10 hour shift is worked, each crew consists of only 30 people. This combination results in requiring 7 days to repair each satellite and consequently a group of 4 crews can only repair 5 satellites during an equinox period. Therefore, in order to repair 10 satellites as was accomplished in option one, another group of 4 crews is required for the repair of the other 5 satellites.



SPS-1916

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## Daily Repair Schedule Options

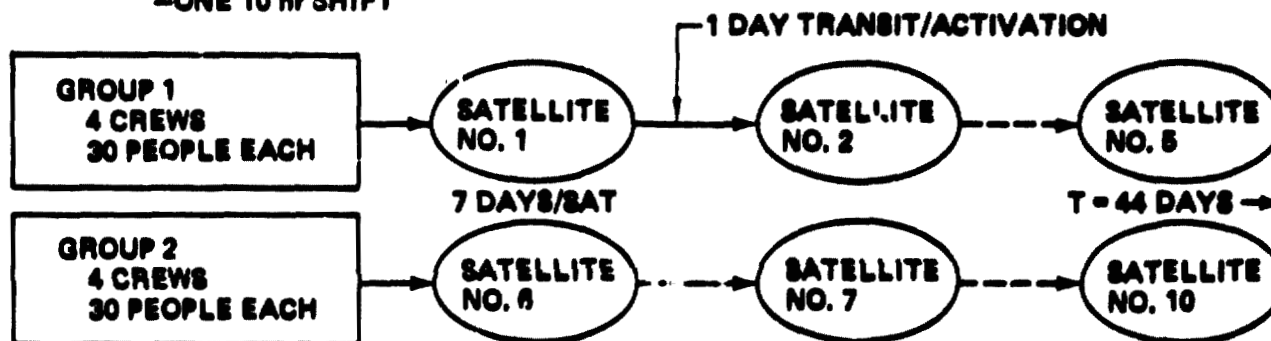
BOEING

### • CONDITIONS

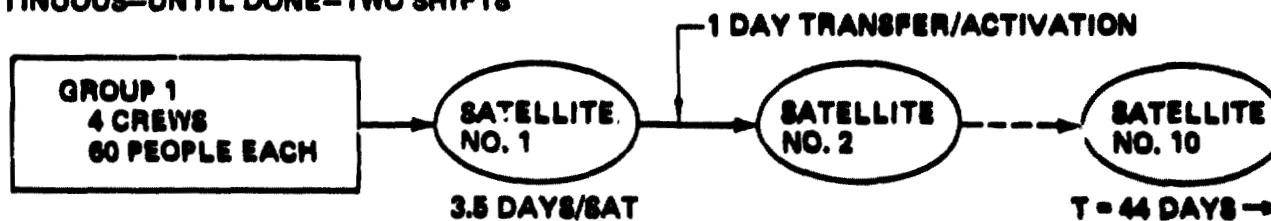
- 2,850 hr OF REPAIR PER SATELLITE
- MAXIMUM OF 4 CREWS PER SHIFT
- 10 REPAIR VEHICLES PER CREW

### • OPTIONS

- INTERMITTANT—CENTER AROUND OCCULTATION  
—ONE 10 hr SHIFT



- CONTINUOUS—UNTIL DONE—TWO SHIFTS

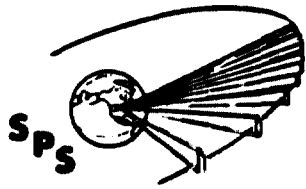




**DAILY REPAIR SCHEDULE COMPARISON**

Comparison of the two daily repair schedule options with 10 satellites in orbit indicates no appreciable difference in cost. The ground grid impact, however, shows that the "continuous" repair case only has one satellite shutdown at a time while the "intermittent" approach requiring two groups working in parallel has two satellites shutdown per unit of time. The effect of the satellite shutdown during a low power period of the day versus continuously shut down during a 3½ day period has not been analyzed at this time.

Further analysis therefore will make use of the "continuous" repair approach.



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## Daily Repair Schedule Selection

● 10 SATELLITES

● ONE EQUINOX

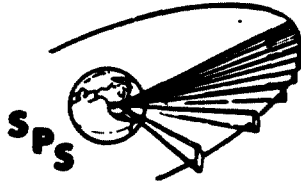
|                                                            | <u>CONTINUOUS</u>                    | <u>INTERMITTANT</u>                |
|------------------------------------------------------------|--------------------------------------|------------------------------------|
| ● DOWNTIME                                                 | 24 hr/DAY<br>2.5 DAY/SAT<br>(\$350M) | 11 hr/DAY<br>7 DAY/SAT<br>(\$330M) |
| ● LOGISTICS                                                | 4 FLIGHTS<br>(\$100M)                | 4 FLIGHTS<br>(\$100M)              |
| ● GROUND POWER GRID<br>SATELLITES DOWN<br>PER UNIT OF TIME | ONE                                  | TWO                                |

SELECTION: CONTINUOUS UNTIL DONE

### **TIME OF YEAR REPAIR OPTIONS**

The time of year comparison is made at a time well into the SPS program where 100 satellites are in orbit. The first of the options called "only during equinox" means all 100 satellites are repaired during the 44-day equinox period. This of course also occurs during the second equinox period when using a semi-annual maintenance concept. As indicated in the time of day analysis, one group of 4 crews can repair 10 satellites in 44 days. Therefore to repair 100 satellites, 10 groups of crews are required to repair the satellites in the designated time.

The second repair option called "equinox to equinox" has the crew beginning to work on the satellite at the start of one equinox and continuing until the start of the next equinox, at which time the cycle is repeated again to satisfy the semi-annual maintenance contract. Working in this manner, a total of 180 days is available to repair the satellites. If each group can repair 20 satellites, a total of 5 groups are required to service the 100 satellites.



D180-24071-3

# Time of Year Repair Options

SPS-1918

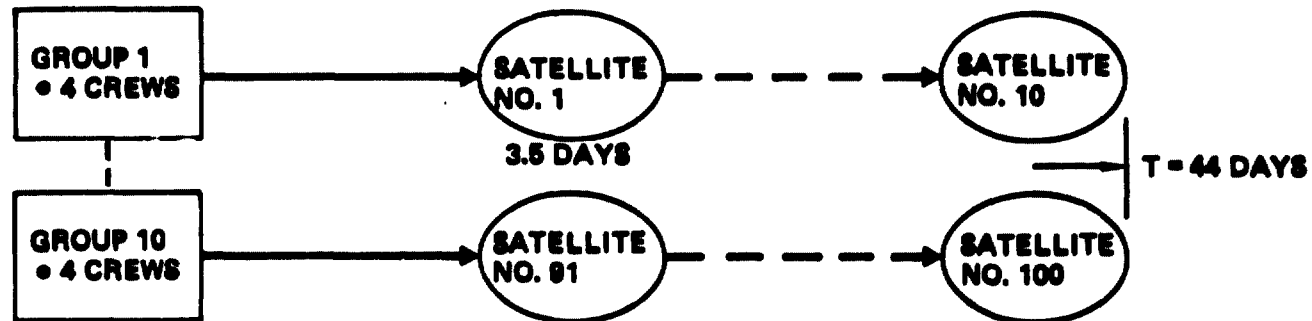
BOEING

## • CONDITIONS

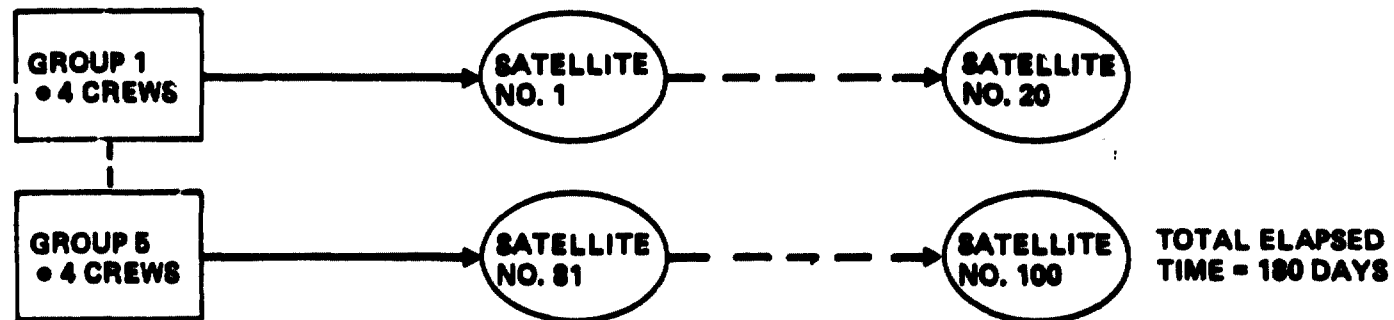
- 100 SATELLITES IN ORBIT
- 4 CREWS/2 SHIFTS
- TRANSFER TIME BETWEEN SATELLITES AND EQUIPMENT ACTIVATION = 1 DAY

## • OPTIONS

- ONLY DURING EQUINOXES



- EQUINOX TO EQUINOX

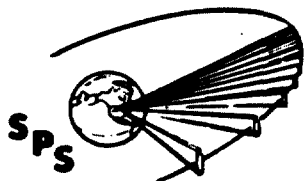


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### **TIME OF YEAR MAINTENANCE SCHEDULE**

The schedule used by the crews in these two options is shown. For the "only during equinox" options, the 10 groups are working in parallel on 10 different satellites and by definition finish the repair work in 44 days. When crew orbital stay times of 90 days are used, the remaining 46 days are spent back at the GEO final assembly base doing klystron tube refurbishment. The second set of 10 groups repeats this operation between days 180 and 270.

The "equinox to equinox" option involves five groups of crews to accomplish the required work. The implementation of this option has three groups repairing satellites the first 90 days in conjunction with two groups stationed at the GEO final assembly base doing klystron tube refurbishment. The second 90 day period has two groups repairing satellites (for a total of five groups in 180 days) along with three groups refurbishing klystron tubes.



D180-24071-3

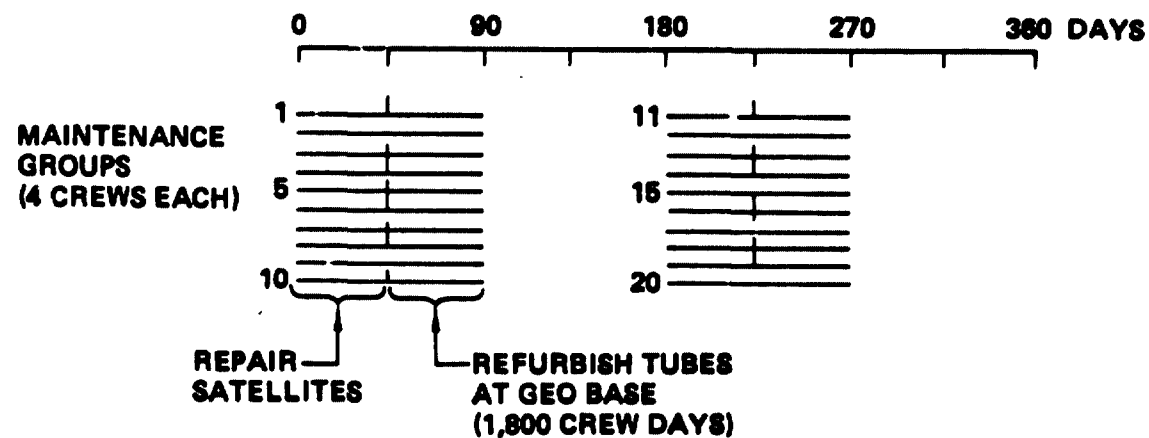
# Time of Year Maintenance Schedule

SPS-1010

**BOEING**

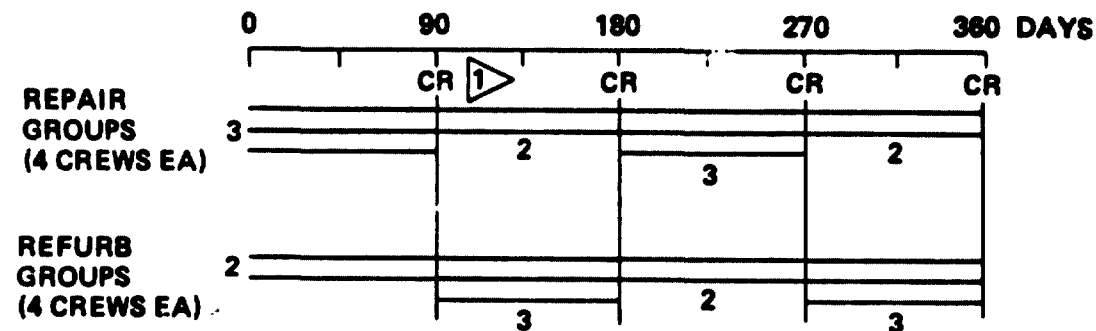
## ● OPTION 1—ONLY DURING EQUINOXES

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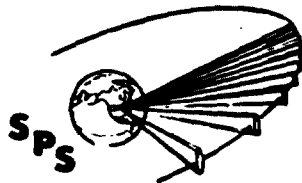
## ● OPTION 2—EQUINOX TO EQUINOX

**1 CREW ROTATION**



**TIME OF YEAR MAINTENANCE SCHEDULE SELECTION**

Comparison of the two maintenance schedules indicates a small cost advantage for the "only during equinox" option for those items have a direct annual cost. Consideration of amortized items, however, give a considerable advantage to the "equinox-to-equinox" option due to fewer crews being in orbit at a given time. This results in a total cost saving of \$1230 million (11%) for the "equinox-to-equinox" option. Another factor highly in favor of the "equinox-to-equinox" maintenance option is the number of satellites shut down per unit of time. Again this advantage is the result of spreading the repair activities out over a longer period of time. The recommendation at this time concerning time of year is that "equinox-to-equinox" maintenance should be utilized.



SPS-1920

D180-24071-3

## Time of Year Maintenance Schedule Selection

**SPRING**

|                                                               | <u>ONLY DURING<br/>EACH EQUINOX</u>                                                                      | <u>EQUINOX TO<br/>EQUINOX</u>                                                                                     |
|---------------------------------------------------------------|----------------------------------------------------------------------------------------------------------|-------------------------------------------------------------------------------------------------------------------|
| ● DOWNTIME<br>(ANNUAL)                                        | <ul style="list-style-type: none"> <li>● 3.38 DAYS/100 SAT</li> <li>● \$6720M</li> </ul>                 | <ul style="list-style-type: none"> <li>● 3.38 DAY/30 SAT</li> <li>● 3.5 DAY/70 SAT</li> <li>● \$6,900M</li> </ul> |
| ● LOGISTICS<br>(ANNUAL)                                       | <ul style="list-style-type: none"> <li>● 40 CREWS</li> <li>● 2 TIMES/YEAR</li> <li>● \$2,000M</li> </ul> | <ul style="list-style-type: none"> <li>● 20 CREWS</li> <li>● 4 TIMES/YEAR</li> <li>● \$2,000M</li> </ul>          |
| ● MAINTENANCE<br>HABITATS<br>(AMORTIZED)                      | <ul style="list-style-type: none"> <li>● 40 UNITS</li> <li>● \$1,440M</li> </ul>                         | <ul style="list-style-type: none"> <li>● 20 UNITS</li> <li>● \$720M</li> </ul>                                    |
| ● REFURB FACILITY<br>AT GEO BASE<br>(AMORTIZED)               | <ul style="list-style-type: none"> <li>● 20 UNITS</li> <li>● \$960M</li> </ul>                           | <ul style="list-style-type: none"> <li>● 6 UNITS</li> <li>● \$270M</li> </ul>                                     |
| ● TOTAL COST DELTA                                            | \$11120M                                                                                                 | \$9890M (11% SAVINGS)                                                                                             |
| ● GROUND GRID IMPACT<br>(SATELLITES DOWN PER<br>UNIT OF TIME) | 10%                                                                                                      | 3%                                                                                                                |

SELECTION: EQUINOX TO EQUINOX



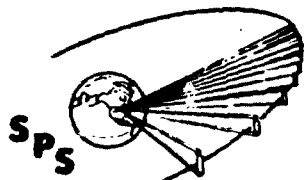
### **MAINTENANCE MISSION OPTIONS**

Several options exist relative to the orbital location of the maintenance system elements and the associated transportation operations. A simplified version of each of these options is presented in this chart, with a transportation comparison following on the next chart.

Option 1 is primarily considered for the case where there are only a few satellites in orbit and consequently a mission approach consisting of maintenance sorties to a satellite similar from a LEO base may be advantageous. In this option, all maintenance elements are located at the LEO construction base except when a repair sortie is performed. Each sortie to GEO has a payload consisting of maintenance habitat plus completely refurbished klystron tube modules.

Option 2 consists of installing as permanent equipment at each satellite the required maintenance habitat and refurbishment facility and equipment. Flights from a LEO base include crews and components to repair the klystron tube modules rather than ready to go klystron tube modules.

Option 3 has the maintenance elements all based at the GEO final assembly base. All maintenance crews and klystron tube module components come to the GEO final assembly base. Maintenance sorties to the satellite are made from the GEO final assembly base and include crew habitats and completely refurbished klystron tube modules. After the satellite is repaired, the sortie returns to the GEO final assembly base with the failed klystron tubes where they will be refurbished with components that have previously been brought up.



0280-24071-3

# Maintenance Mission Options

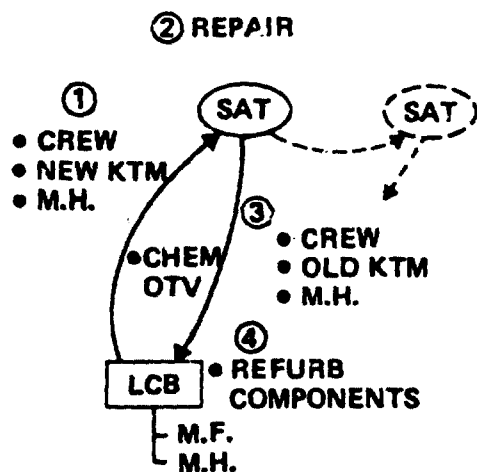
SPS-1924

BOEING

## • KEY ISSUES

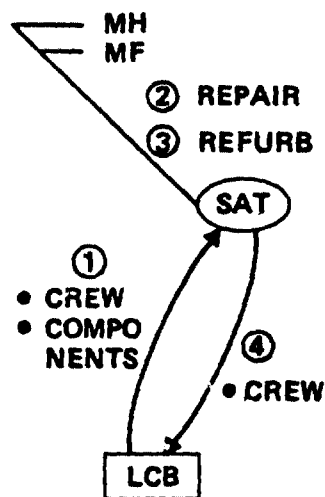
- MAINTENANCE HABITAT LOCATION
- MAINTENANCE FACILITY (REFURB) LOCATION
- TRANSPORTATION

### OPTION 1—LEO BASED

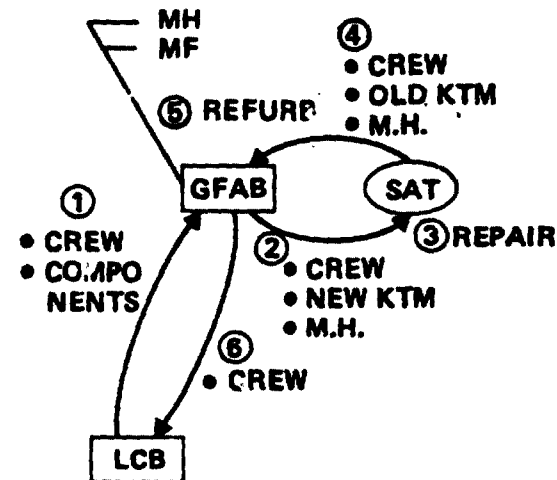


- LCB = LEO CONST BASE
- GFAB = GEO FINAL ASSEMBLY BASE

### OPTION 2—SAT BASED



### OPTION 3—GEO BASED



- M.H. = MAINTENANCE HABITAT
- M.F. = MAINTENANCE FACILITY
- KTM = KLYSTRON TUBE MODULES

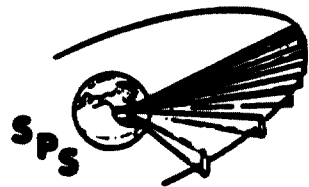
### TRANSPORTATION REQUIREMENTS—MISSION OPTIONS

C-3  
Comparison of the mission options for their transportation requirements in terms of payload mass and delta V indicates Option 3 using GEO based systems the most desirable.

Option 1 with the maintenance system based at LEO has both high mass since habitats and complete klystron modules must be transported and high delta V since these two elements must be brought back to the LEO base.

Option 2 with the maintenance system permanently attached to satellites only requires components which have a mass 38% as much as complete klystron tube modules to be delivered to GEO. Although this option is good from the transportation standpoint, installation of the habitats and refurbishment facilities at each satellite presents an unacceptable investment cost.

Option 3 also only delivers klystron components to GEO (after the habitats and refurb facilities are initially delivered) and then transfers the habitat and refurbished klystron tube modules from one satellite to another which involves very small delta V requirements.



SPS-1902

D180-24071-3

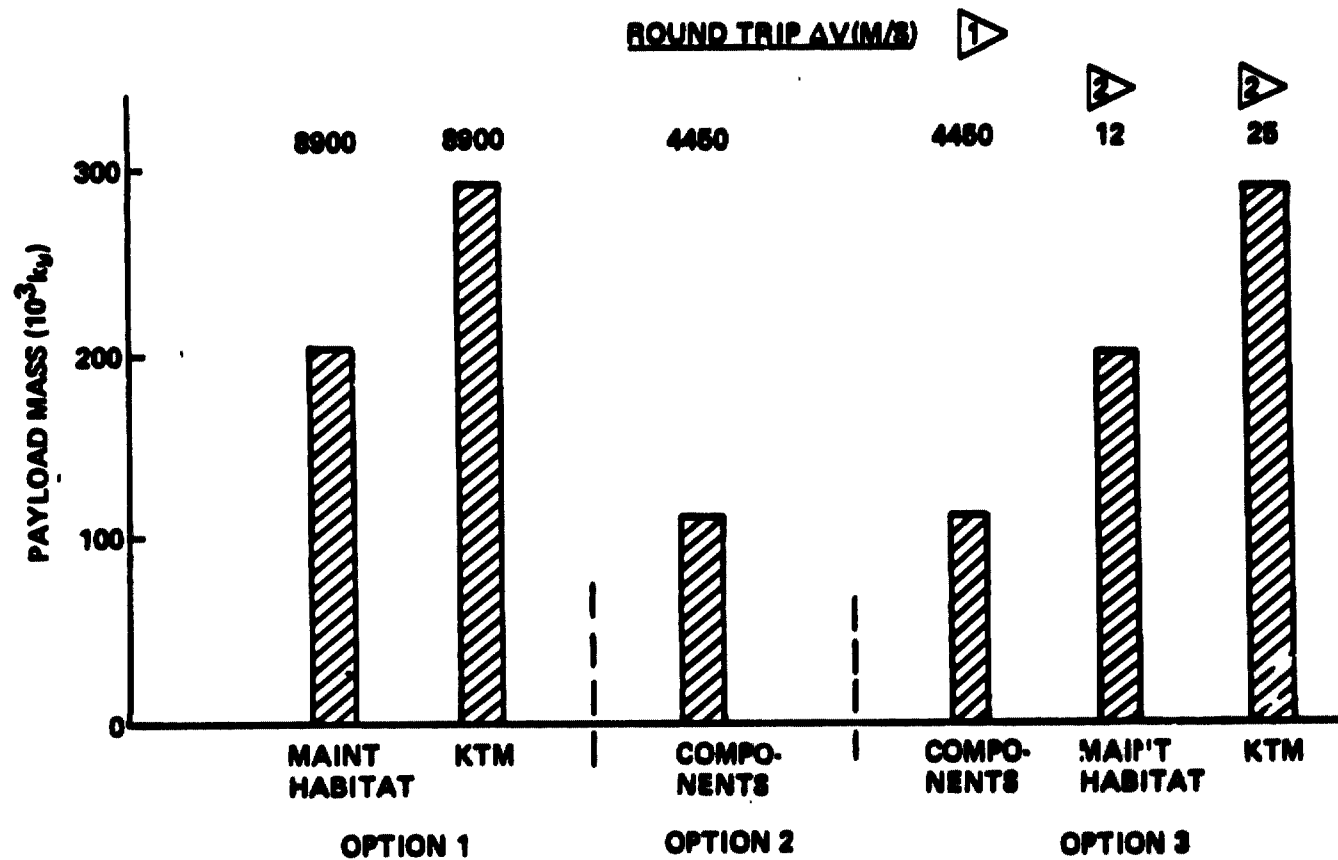
# Transportation Requirements Mission Options

BEING

1 LEO/GEO EXCEPT AS NOTED

2 GEO/GEO

ROUND TRIP AV(M/S)



KTM = KLYSTRON TUBE MODULE

### **SELECTED MAINTENANCE MISSION CONCEPTS**

The final operations associated with a typical 90-day period are illustrated. After the 20th satellite has been repaired, the crew and habitat return to the GEO final assembly base where the habitat is left for the next repair crew. The initial crew then returns back to the LEO construction base and eventually back to Earth. The refurbishment crew has also completed their 90 day stay time and also returns back to Earth.

Four new repair crews and four new refurbishment crews are then transferred to the GEO final assembly base. The complete cycle is repeated again. The crew size at the GEO final assembly base will have a maximum operating size of 310 (240 associated with refurbishment and 70 with satellite assembly) and at the time the four repair crews return at the end of their tour of duty the crew size will be 550.

The annual number of orbit transfer vehicles and launch vehicles flights which occur in the maintenance of 100 satellites are also indicated. During this time period, maintenance operations will completely dominate the GEO transportation operations rather than assembly of satellites at GEO.



## Selected Maintenance Mission Concept

**• SPRING**



- ΔΔΔΔ

### **SELECTED MAINTENANCE MISSION CONCEPT**

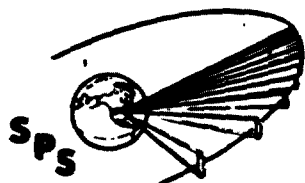
The selected maintenance mission concept is now presented in greater detail in the next two charts. The operations discussed in these charts will be those associated with one GEO final assembly base and the operations associated with one repair group and one refurbishment group. Other final assembly bases would have comparable operations.

Once maintenance operations are begun, the GEO construction base serves as a major staging depot for the maintenance crews and their hardware in addition to its role of constructing the satellites. Typically, the following operations occur. Four repair crews and four refurb crews are transported to the GEO final assembly base. Each crew is provided with its own orbit transfer vehicle. At approximately the same time another orbit transfer vehicle delivers klystron tube module components to be used in the refurbishment of failed tubes.

Refurbishment crews remain at the GEO final assembly base, repairing the failed klystron tube modules that have previously been delivered by other repair crews. Repair crews transfer to the satellite designated for repair taking with them their habitat. The second stage of the orbit transfer vehicle which brought the crew to GEO is used for the transfer to the satellite. The second stage of the orbit transfer vehicle used to deliver the klystron tube components to the GEO final assembly base is then loaded with refurbished klystron tube modules and transferred to the first satellite to be repaired.

At the completion of repairs on the first satellite, the crew and habitat transfer to the next satellite to be repaired. The other orbit transfer vehicle transport the failed klystron tube modules back to the GEO final assembly base where they will be refurbished. The OTV then returns back to the LEO construction base. Prior to this time, however, another orbit transfer vehicle has come from the LEO construction base to the GEO final assembly base delivering additional klystron tube components and is then dispatched with completely refurbished klystron tube modules to the second satellite that is to be repaired.

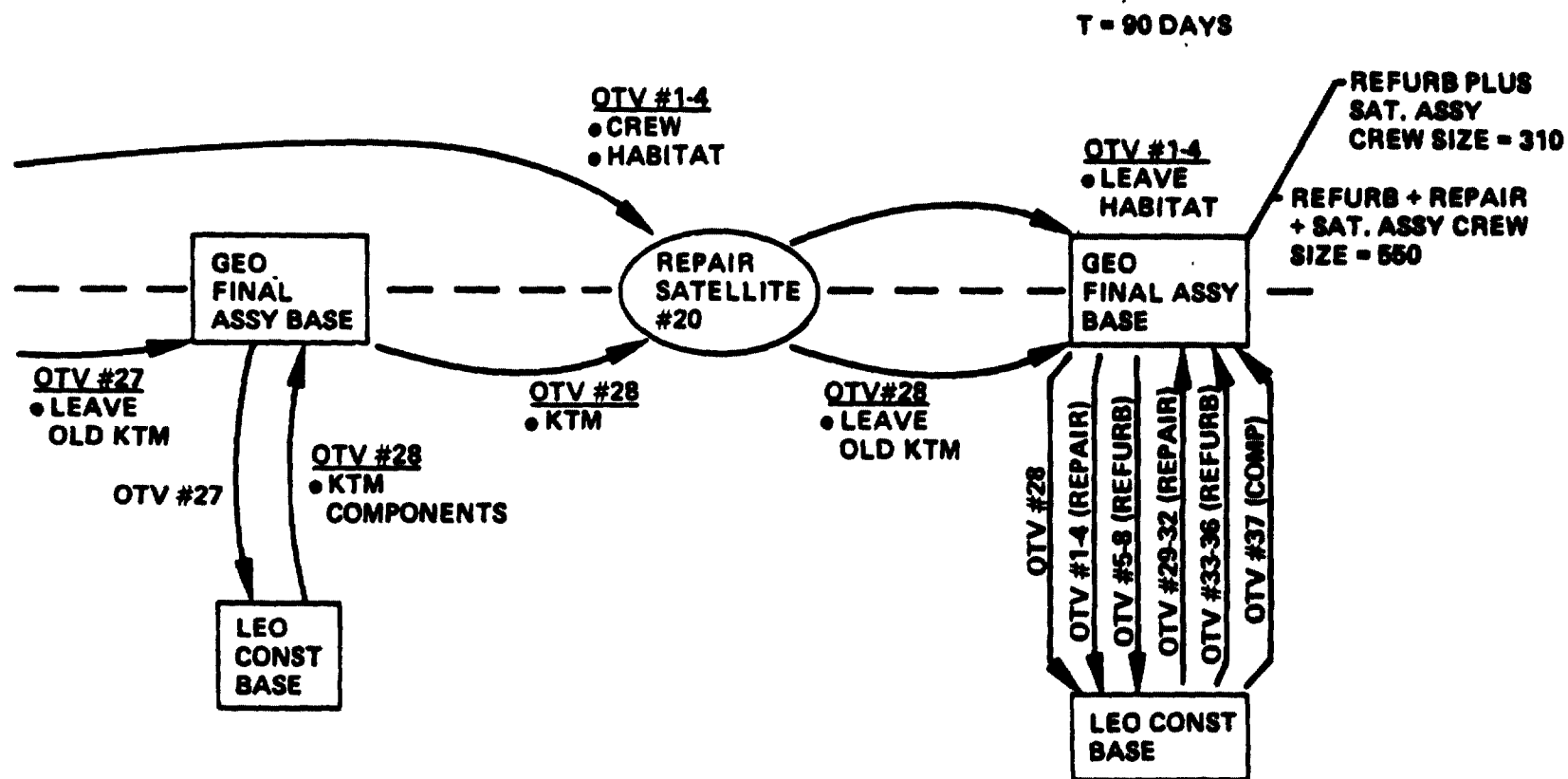
This cycle is repeated for each satellite to be repaired.



SPS-1836

# Selected Maintenance Mission Concept

ORION



## ANNUAL FLT SUMMARY (100 SATELLITES)

- 280 OTV FLTS TO GEO
- 400 OTV FLTS GEO TO GEO
- 350 HLLV FLTS TO LEO
- 80 SHUTTLE GROWTH FLTS TO LEO



**SATELLITE MAINTENANCE SYSTEMS**

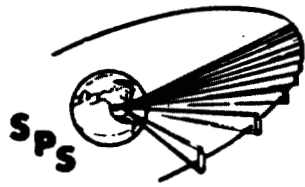
The satellite maintenance systems and their location on an antenna are shown on the next two charts. The systems are shown as they relate to one side of one antenna. Since four crews work on each satellite, those same systems are present on both sides of both antennas.

To enable the docking of the various maintenance system elements and to transfer cargo around the antenna, the antenna structure has been designed to incorporate a cargo distribution system and has structural additions to allow maintenance gantries to be positioned so they can be maintained and supplied with new klystron tube modules.

The 60 person crew is delivered to the satellite in the crew habitat using the second stage of the OTV that initially brought the crew from the LEO construction base to the GEO final assembly base. Once at the satellite (antenna), a crew bus is used to transfer persons between the habitat and the maintenance repair vehicles.

Cargo, primarily in the form of klystron tube modules is also delivered to the satellite using a dedicated OTV (stage 2) that had initially brought klystron components to the GEO final assembly base for refurbishment of "failed" klystron tubes. The operations associated with an OTV include docking and release of one klystron tube pallet on one side of the antenna and then free-flying to the other side of the antenna leaving another pallet followed by flying to the other antenna and leaving two pallets in a similar manner. At the completion of the repair operation, the pallets are loaded with "failed" klystron tubes. The OTV then moves to the four docking locations collecting the pallets and then returns them back to the GEO final assembly base where they will be refurbished. Following the release of the pallets, the OTV returns to the LEO construction base where it is made ready to deliver another load of klystron components.

The actual distribution of the cargo around the antenna is accomplished through use of cargo transporters operating on the track system on two sides of each antenna. The cargo transporter system consists of three separate units attached together to form a "train". The middle unit is a control unit that has a crew cabin, power systems and crane/manipulator that moves the cargo between the train and the maintenance gantries. Units on either side of the control unit are essentially trailers that carry either new klystron tube modules or those that have failed and have been removed. The train system moves down to each gantry and delivers to it the number of klystron tubes required in that particular antenna channel during one shift or one day of operation depending on the channel.

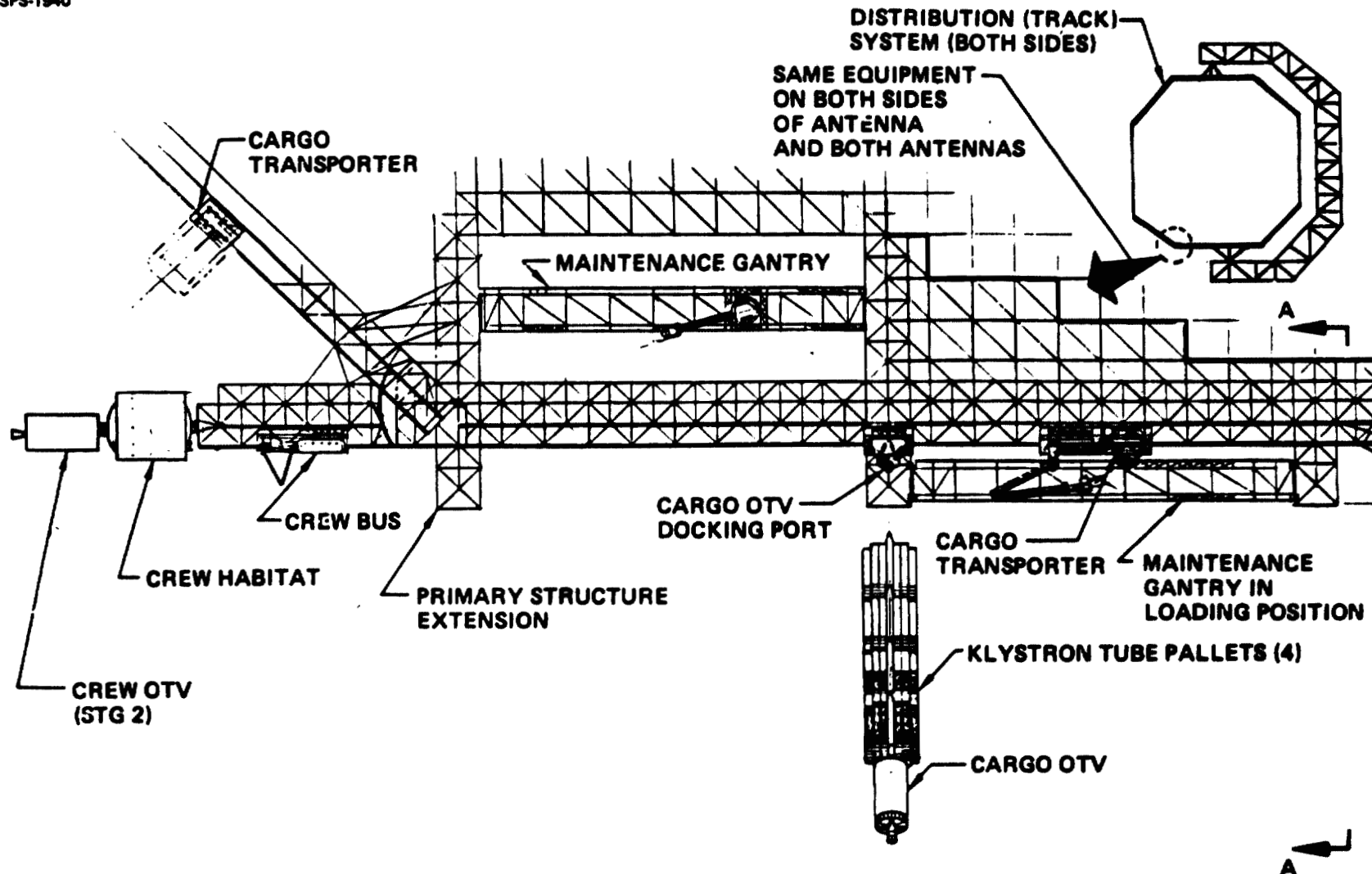


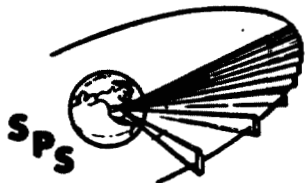
SPS-1940

D180-24071-3

# Satellite Maintenance Systems

BOEING



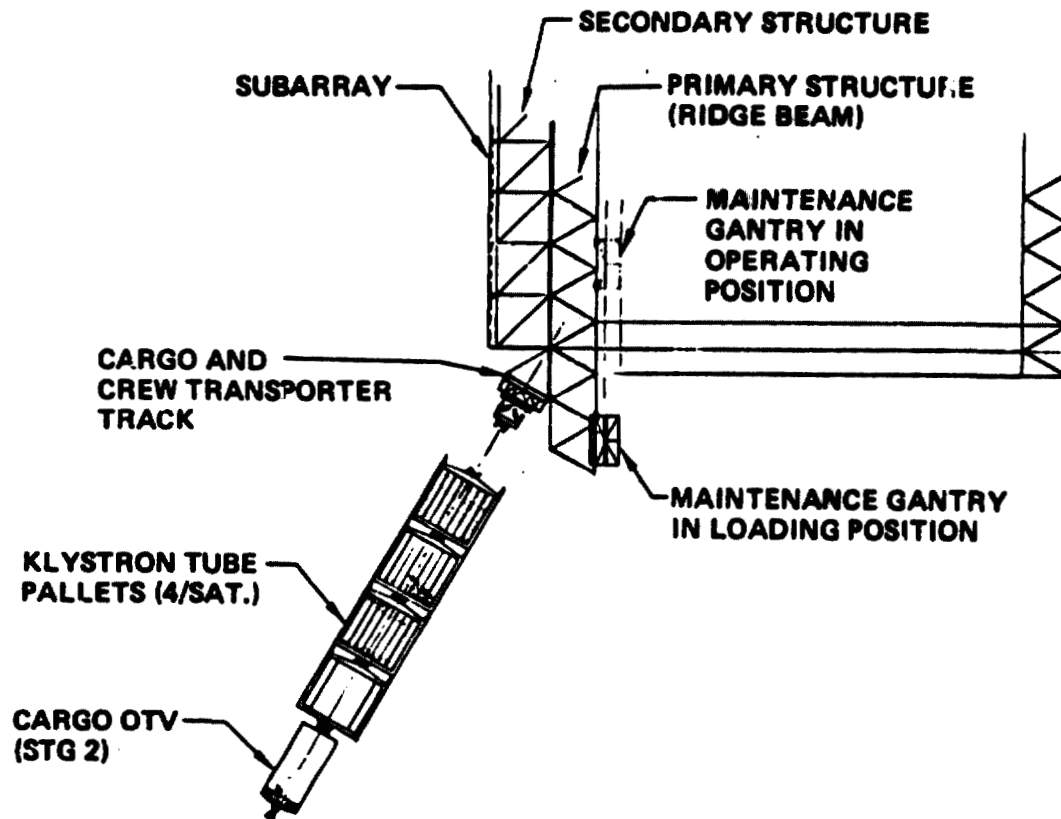


SPS-1941

D180-24071-3

# Satellite Maintenance Systems

BOEING



A-A

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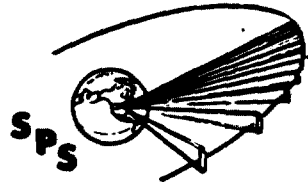
### **ANTENNA MAINTENANCE SYSTEM INSTALLATION**

The installation of the maintenance equipment on an antenna being repaired by two crews is shown.

The number of maintenance vehicles (machines) installed in each channel of the antenna is a function of the estimated number of tube failures. This value is larger in the middle channels of the antenna since the center of the antenna has subarrays containing 36 and 30 klystron tube modules while near the edge of the antenna, the subarrays have 4 or 6 tubes per subarray. Consequently it will be noted that the middle channel has three maintenance systems consisting of a gantry and repair vehicle.

With this equipment distribution and working 20 hours per day, the middle channels require slightly more time than previously identified for repair—3 1/2 days per satellite. The addition of 1/2 day to the schedule, however, will not appreciably alter the prior analysis.

It should also be noted, the outside channels require far less time to repair and less equipment due to fewer failed tubes. Consequently when the crews assigned to this particular equipment are finished, they can then be used to repair other components on the satellite such as the dc-dc converters mentioned earlier in the discussion.



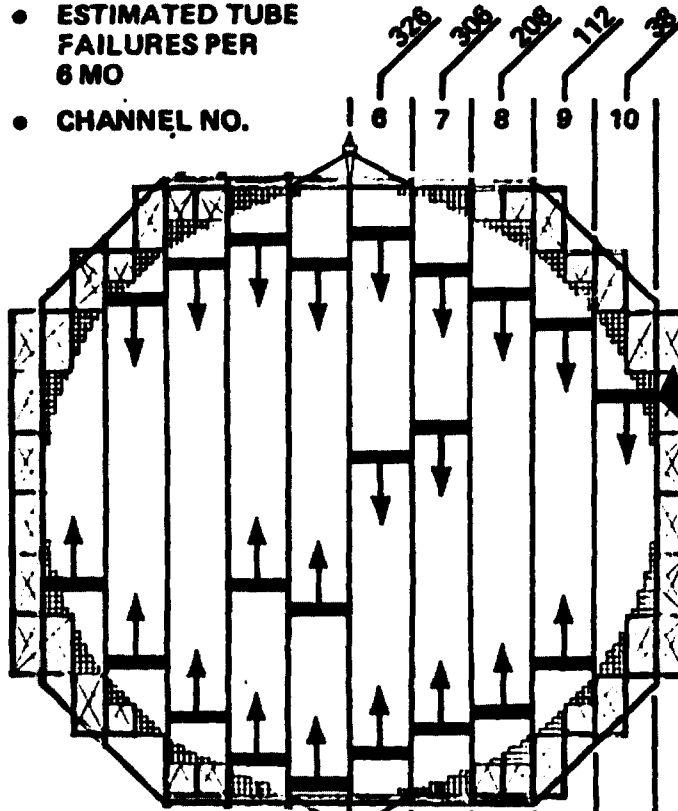
D180-24071-3

# Antenna Maintenance System Installation

SPS-1837

BOEING

- ESTIMATED TUBE FAILURES PER 6 MO
- CHANNEL NO.



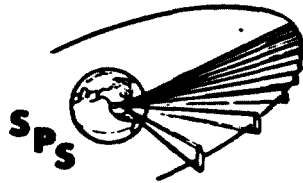
|                   |        |     |     |     |     |
|-------------------|--------|-----|-----|-----|-----|
|                   | GIMBAL |     |     |     |     |
| • NO. OF MACHINES | 3      | 3   | 2   | 2   | 1   |
| • DAYS TO REPAIR  | 4.0    | 3.8 | 3.9 | 2.1 | 1.4 |
| 24 hr DAY         | 3.4    | 3.2 | 3.3 |     |     |

20 hr/DAY  
3.5 DAY ALLOWABLE

**ANNUAL OPERATING CHARACTERISTICS**

The plant factor resulting from the selected maintenance approach is 0.92. The occultation loss cannot be prevented. The output degradation loss is the average which results from using a semi-annual maintenance approach. Downtime loss is the 3-1/2 to 4 day repair time per satellite occurring twice a year. A 3% allowance for such things as unexpected failures and unknowns is also included.

The total cost to maintain 100 satellites per year is over \$11 billion or approximately \$118 million per satellite. It should be noted that some of these costs are annual costs such as transportation (delivery) while others are amortization of the maintenance equipment. Although the maintenance cost may appear high it still is less than 3% of the yearly revenue per satellite and can be expressed as \$0.0015 per kilowatt hour.



SPS-1933

D180-24071-3

# Annual Operating Characteristics

**BEING**

## PLANT FACTOR LOSSES

|                                         |             |
|-----------------------------------------|-------------|
| ● OCCULTATION                           | 1%          |
| ● OUTPUT DEGRADATION<br>DUE TO FAILURES | 2%          |
| ● PLANNED MAINTENANCE<br>DOWNTIME       | 2%          |
| ● ALLOWANCE FOR OTHER<br>FACTORS        | 3%          |
| <b>ESTIMATED FACTOR</b>                 | <b>0.92</b> |

## MAINTENANCE COST/YEAR (100 SATELLITES)

|                                    |                  |
|------------------------------------|------------------|
| ● CREW DELIVERY                    | \$2,000M         |
| ● KTM DELIVERY                     | \$3,385M         |
| ● KTM COST                         | \$1,520M         |
| ● MAINTENANCE HABITAT <sup>1</sup> | \$720M           |
| ● REFURB FACILITY <sup>1</sup>     | \$270M           |
| ● REPAIR EQUIPMENT <sup>1</sup>    | \$2,400M         |
| ● REPAIR EQUIPMENT DELIVERY        | \$1,490M         |
| <b>TOTAL</b>                       | <b>\$11,785M</b> |
| <b>AVG/SAT</b>                     | <b>\$118M</b>    |

(3% OF YEARLY REVENUE)

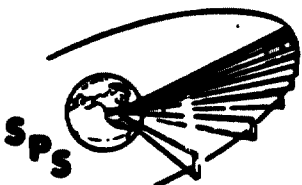
$$\begin{aligned}
 \bullet \text{ MAINTENANCE COST } (\$/\text{kWh}) &= \frac{\text{ANNUAL MAINT COST/SAT}}{8,766 \text{ HR} \times \text{PLT FACTOR} \times 10^7 \text{ kW}} \\
 &= \frac{\$118 \times 10^6}{8,766 \times 0.92 \times 10^7} = 0.0015 \blacktriangleleft
 \end{aligned}$$

<sup>1</sup> AMORTIZED (15% CAPITAL CHARGE)

### **MAINTENANCE AND OPERATIONS SUMMARY**

An overall summary of the selected maintenance approach for each of the key issues mentioned at the beginning of the analysis is presented. Although these selections should not be considered as the most optimum they do present a reasonable approach from the standpoint of both cost and practicality. In summary, maintaining solar power satellites does not present an overwhelming obstacle as some had initially envisioned.





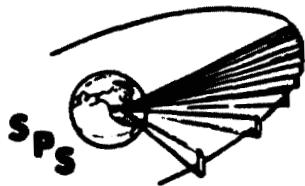
D180-24071-3

## Maintenance and Operations Summary

SP-1823

BEING

- KEY CONCERN.....KLYSTRON TUBES
- LEVEL OF REPLACEMENT..... TUBE MODULE
- REPAIR CONCEPT.....VERTICAL ACCESS  
CUBIC SECONDARY STRUCTURE  
"A" FRAME PRIMARY STRUCTURE
- FREQUENCY OF MAINTENANCE .....SEMI-ANNUAL
- MAINTENANCE SCHEDULE .....EQUINOX TO EQUINOX  
DAILY UNTIL DONE
- CREWS/SATELLITE .....FOUR
- MAINTENANCE HABITAT.....BASE AT GFAB  
MOVE TO SATELLITE
- REFUBRISH LOCATION .....AT GFAB
- LEO-GEO TRANSPORTATION .....CREW—>CHEM OTV  
...COMPONENTS—>CHEM OR SAT MODULE
- GEO-GEO TRANSPORTATION .....ALL ITEMS VIA CHEM OTV
- COST/SATELLITE.....LESS THAN \$120 MILLION  
...LESS THAN 2 MILL/KWHR



SP4-1878

D180-24071-3

## MPTS Studies

**BOEING**

- ARRAY ERROR ANALYSIS
- ALTERNATE FREQUENCY STUDIES
- PHASE CONTROL
- RECTENNA

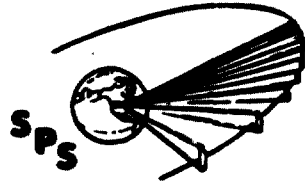
**E. Nalos**

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**ARRAY ERROR ANALYSIS**

**GRATING LOBE STUDIES**

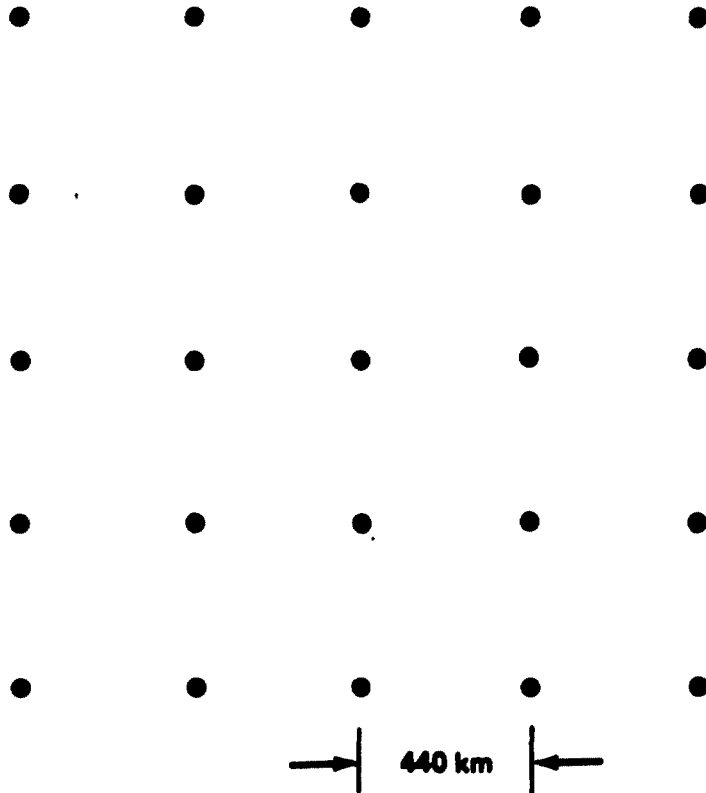
The grating lobe positions along the principal planes are determined by the following equation:  $X = R \tan(\sin^{-1} \frac{n\lambda}{L+gap})$  where  $R$  is the orthogonal range,  $n$  is the order of the grating lobe,  $L$  is the subarray length, and  $gap$  is the spacing between subarrays. In this study only the lobes along the principal planes have been studied. The computer program is presently being modified to study grating lobe behavior at any  $\phi$  cut desired. Systematic tilt has the major effect on the grating lobe levels (over random subarray tilt, power taper, quantization, subarray spacing and other errors) due to the shifting of the grating lobe peaks out of the nulls of the subarray pattern.



SPS-1280

BOEING

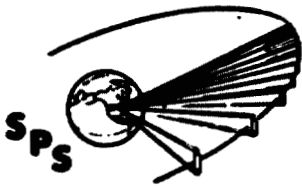
## Grating Lobe Positions



- 10 x 10m SUBARRAYS HAVE GRATING LOBES SEPARATED BY 440 km AT 2.45 GHz
- WIDTH OF UNSPLIT GRATING LOBES CORRESPOND TO  $\approx$  13 km, SAME AS MAIN BEAM.
- SYSTEMATIC TILT HAS MAJOR EFFECT ON GRATING LOBE LEVELS.

**GRATING LOBE PEAKS PRODUCED BY SYSTEMATIC SPACETENNA TILT**

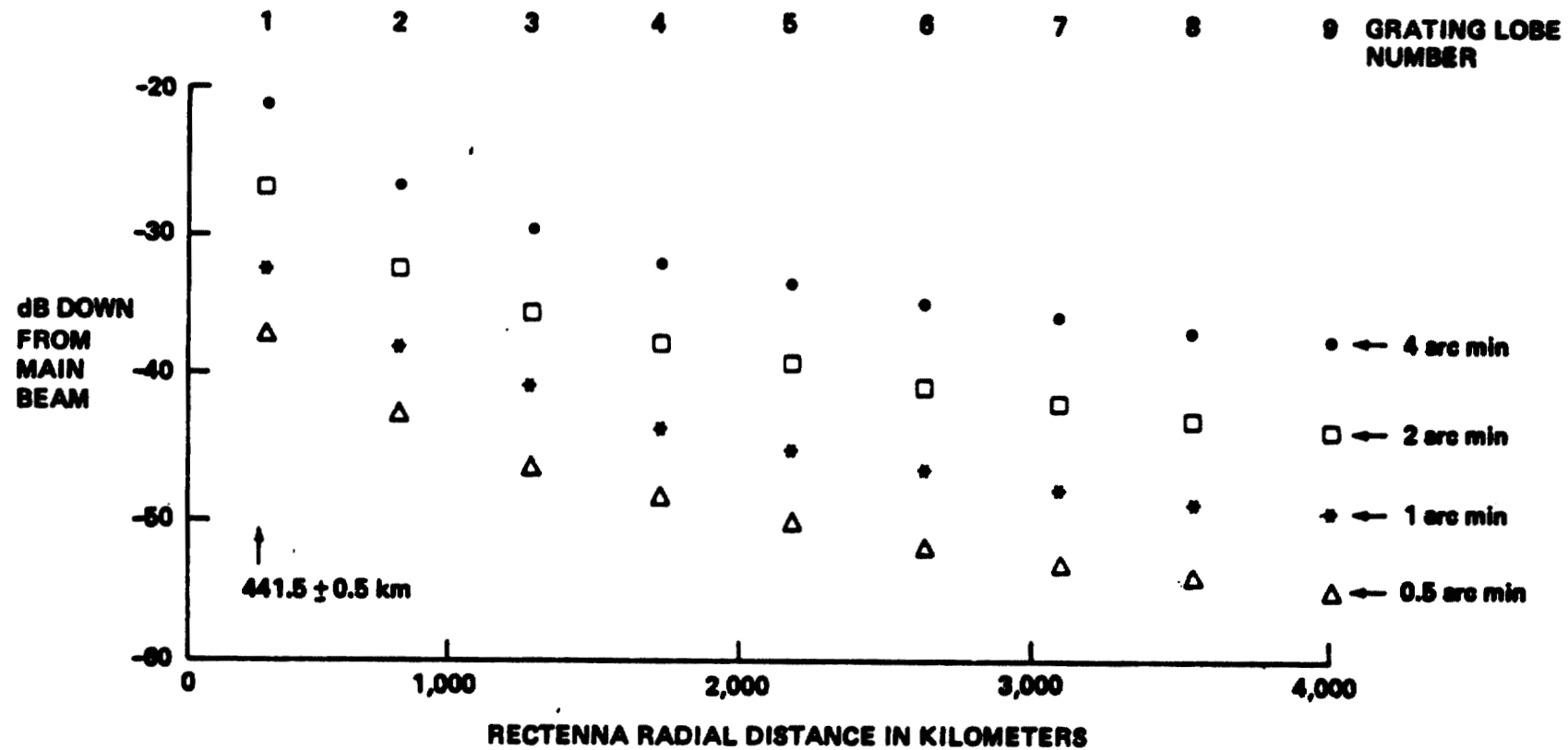
Nine grating lobe peaks are shown for each of four cases of systematic spacetenna tilt. As a simple rule-of-thumb for each doubling of systematic tilt the grating lobe amplitudes increase approximately six decibels. It is important to note that the grating lobe levels may be larger than the first sidelobe ( $\approx 23.5$  dB down) as shown for the first grating lobe for the four arc-minute tilt case. The grating lobes roll off at approximately 22 dB per decade.



SPS-1870

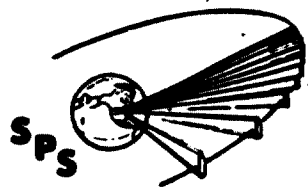
BOEING

# Grating Lobe Peaks Produced by Systematic Spacenna Tilt



**EFFECT OF RANDOM TILT ON GRATING LOBE LEVEL**

Two cases of random tilt are shown, one and four arc minutes, for three of the nine grating lobes studied. The random tilt is incorporated in the phased array computer program by adding a random angle to theta, utilizing the Gaussian random number generator and the specified standard deviations of one or four arc minutes.

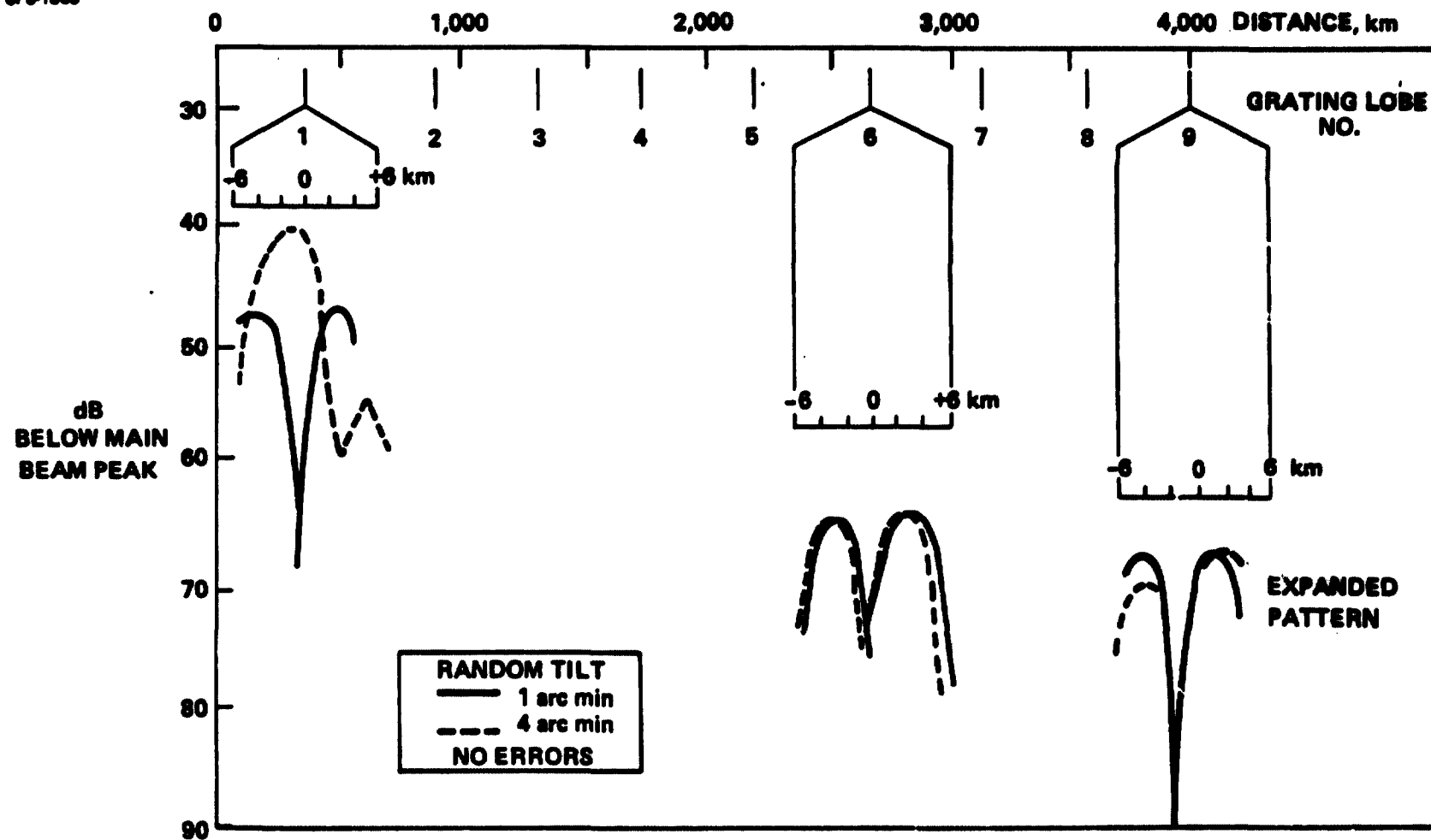


D180-24071-3

# Effect of Random Tilt on Grating Lobe Level

SPS-1840

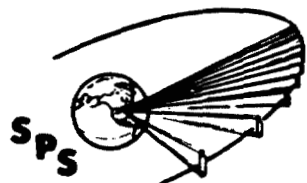
BOEING





**SPACETENNA PATTERN ROLL-OFF CHARACTERISTICS**

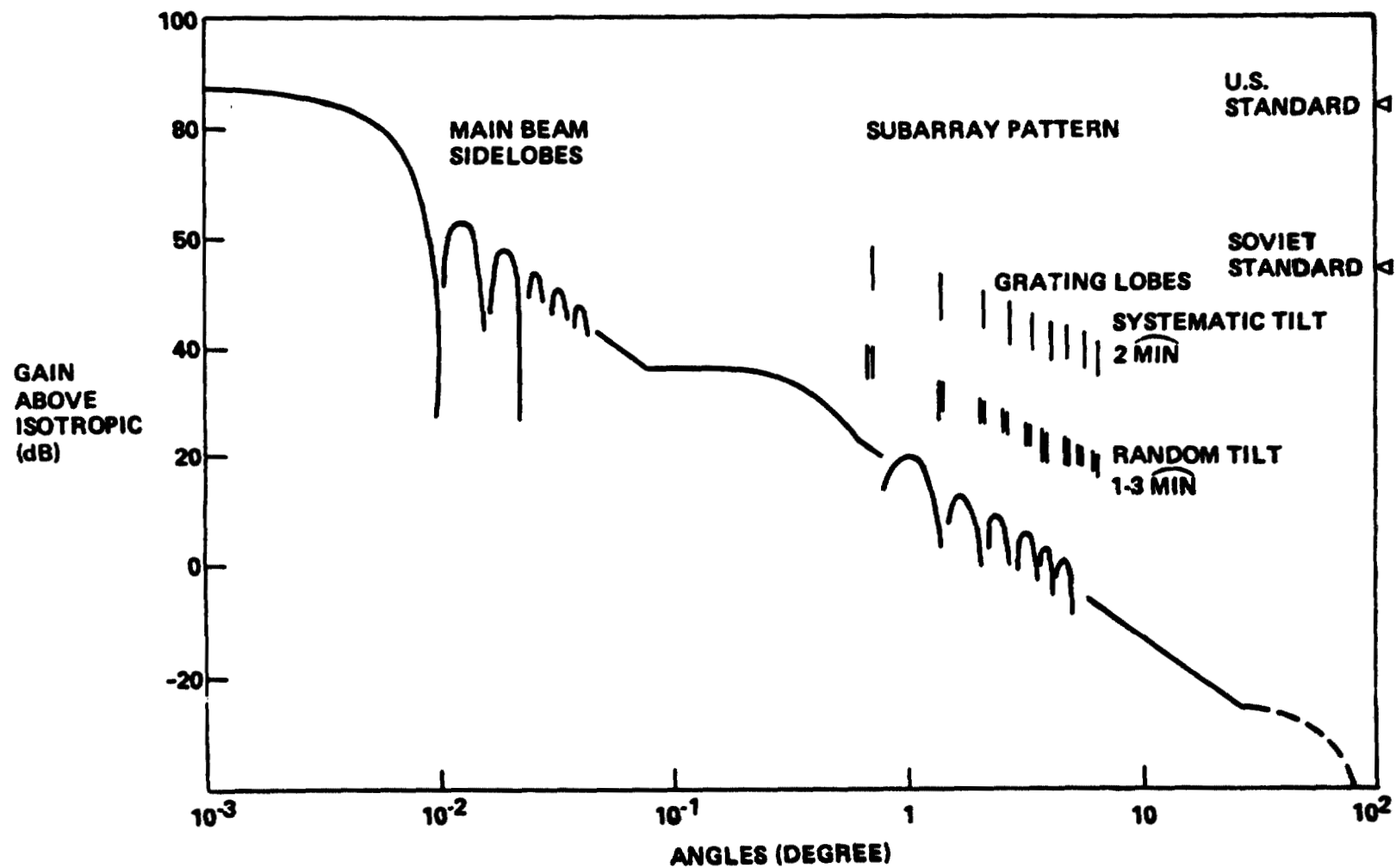
The grating lobe levels are shown for both systematic and random tilt. For the case of systematic tilt, the grating lobe peaks are moved out of the nulls of the subarray pattern. For random tilt, the grating lobes are split because statistically the peaks lie in the nulls of the subarray pattern.

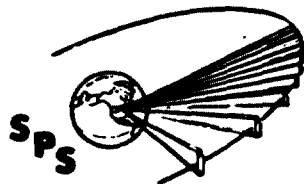


SPS-1885

BOEING

# Array Pattern Roll-Off Characteristics





SPS-1864

D180-24071-3

## Grating Lobe Effects

BOEING -

### SUBARRAY TILT

SYSTEMATIC TILT IS THE MAJOR FACTOR AND NEEDS TO BE MINIMIZED. RANDOM TILT HAS LITTLE EFFECT FOR NOMINAL VALUES.

### QUANTIZATION

THERE IS NO DIFFERENCE ON THE GRATING LOBE AMPLITUDES BETWEEN THE CONTINUOUS GAUSSIAN DISTRIBUTION AND THE 10 STEP QUANTIZED DISTRIBUTION.

### ILLUMINATION TAPER

AS TAPER DOUBLES FROM 10 TO 20 DB THE GRATING LOBE AMPLITUDES INCREASE BY 0.1 DB

### SUBARRAY SPACING

FOR A UNIFORM OR RANDOM  $\frac{1}{4}$ " SUBARRAY GAP SPACING (1.65 ARCSEC), THERE IS NO SIGNIFICANT CHANGE IN GRATING LOBE LEVEL.

### OTHER ERRORS

INCORPORATING THE NOMINAL ERRORS OF: 10° RANDOM PHASE ERROR, 1DB RANDOM AMPLITUDE ERROR, AND 2% FAILURES PRODUCED NO CHANGE IN GRATING LOBE AMPLITUDES. INTRA SUBARRAY ERRORS NEED FURTHER STUDY.

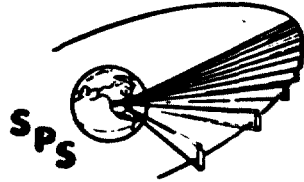
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**KLYSTRON DESIGN FOR 5.8 GHZ**

The klystron scaling to higher frequencies is relatively straightforward; however, a conservative power level for 30 year life needs to be further assessed. The assumptions made are: the same rf power level of 70 kw, same voltage limit of 42 kv, 5 segment depressed collector, output gap thermal stress of  $.5 \text{ kw/cm}^2$ , solenoid focusing and estimated collector recovery of 54%. The various beam and efficiency parameters are listed in the following table.

The scaling relationships are illustrated in the attached figure and reflect a frequency factor of 2.37. The length of the rf section is reduced by 2.37 for the same transit time, the solenoid power increases by 2.37 to a 3.5 kw design for 300°C operation, cavity Q's are reduced by  $1/\sqrt{2.37}$ , resulting in an estimated rf efficiency reduction of 1.62% with a total efficiency reduction of 3.59% including the increased solenoid power.

The mass reduction, indicated in the following table, is estimated at 27%, primarily due to the shorter, lighter solenoid. If an equal weight were selected, the solenoid power could be reduced and the efficiency reduction would drop from 3.59% to 2.61%.



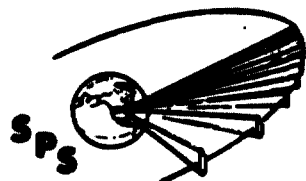
SPS-1876

D180-24071-3

# Alternate Frequency Assessment

**BOEING**

- KLYSTRON EFFICIENCY ANALYSIS
- PARAMETRIC ACQUISITION COST ANALYSIS
- FREQUENCY OPTIMIZATION



D180-24071-3

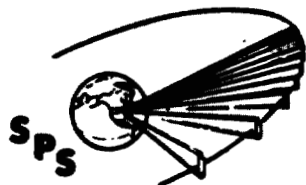
## Klystron Design Parameters Versus Frequency

SPS-1882

BEING

| EFFICIENCY PARAMETER |                     | 2.45 GHz | 5.8 GHz |
|----------------------|---------------------|----------|---------|
| CIRCUIT              | $\eta_{\text{cct}}$ | 0.967    | 0.949   |
| ELECTRONIC           | $\eta_e$            | 0.769    | 0.768   |
| COLLECTOR            | $\eta_c$            | 0.54     | 0.54    |
| TOTAL                | $\eta_{\text{tot}}$ | .850     | .834    |
| UNDEPRESSED          | $\eta$              | 0.745    | 0.728   |

|                            | 2.45 GHz  | 5.8 GHz   |
|----------------------------|-----------|-----------|
| RF POWER                   | 70 kW     | 70 kW     |
| BEAM POWER                 | 93.96 kW  | 96.15 kW  |
| VOLTAGE                    | 42 kV     | 42 kV     |
| CURRENT                    | 2.24 AMPS | 2.29 AMPS |
| PERVEANCE $\times 10^{-6}$ | 0.260     | 0.266     |
| COLLECTOR RECOVERY         | 11.6 kW   | 12.20 kW  |
| NET BEAM POWER             | 82.3 kW   | 83.95 kW  |
| RF EFFICIENCY              | 85.0%     | 83.36%    |
| SOLENOID POWER             | 1.4 kW    | 3.5 kW    |
| NET EFFICIENCY             | 83.63%    | 80.04%    |



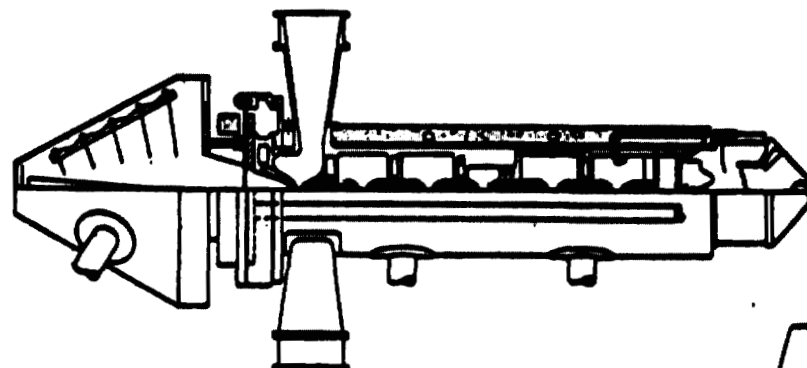
SPS-1981

D180-24071-3

# Approximate Scaling Considerations for High Power SPS Klystron

BOEING

5.8 GHz



REFOCUSING SOLENOID  
INTERNAL COLLECTOR  
HEATPIPE/EVAPORATORS  
COLLECTOR  
PLATES

CAVITY/SOLENOID HEATPIPE  
EVAPORATORS

MAIN SOLENOID

RF INPUT FROM  
SOLID STATE  
CONTROL DEVICE

MOD  
ANODE

2.45 GHz

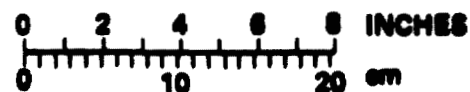
COLLECTOR  
HEATPIPE (2)  
TO RADIATOR

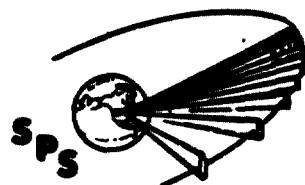
OUTPUT WAVEGUIDE

CAVITY/SOLENOID HEATPIPES (4)  
TO THERMAL RADIATORS

HEATER

CATHODE





D180-24071-3

# Klystron Mass Estimate

SPS-1883

SEEING

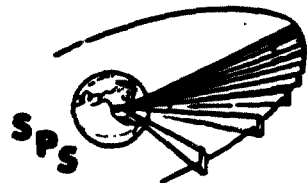
| ITEM                    | 2.45 GHz (REF. 2) |            | 5.8 GHz      |                |            |
|-------------------------|-------------------|------------|--------------|----------------|------------|
|                         | WEIGHT (kg)       | POWER (kW) | SCALE FACTOR | WEIGHT (kg)    | POWER (kW) |
| <b>SOLENOID (300°C)</b> |                   |            |              |                |            |
| 1000 GAUSS              | 20                | 1.4-1.8    |              |                |            |
| 3-in ID, 4-½ in OD      |                   |            |              |                |            |
| 18.5 in LONG            |                   |            |              |                |            |
| 2,370 GAUSS             |                   |            | 1.2          | 8.34           |            |
| 1-¾ in ID, 3-½ in OD    |                   |            | 1.1          |                | 3.5        |
| 6.98 in LONG            |                   |            |              |                |            |
| <b>BODY</b>             |                   |            |              |                |            |
| 18.5-in LONG            | 10                |            |              |                |            |
| 6.98-in LONG            |                   |            |              | 6.5            |            |
| <b>COLLECTOR</b>        | 7                 |            |              | 5.4            |            |
| <b>RADIATOR</b>         |                   |            |              |                |            |
| BODY AT 300°C           | 9.5               |            | 1.2          | 11.4           |            |
| COLLECTOR AT 500°C      | 4.9               |            | 1.2          | 5.9            |            |
| <b>TOTAL</b>            | <b>51.4 kg</b>    |            |              | <b>37.5 kg</b> |            |



### END-END EFFICIENCY DEGRADATION FACTORS

To arrive at a dollar revenue from the grid, it is necessary to obtain a parametric expression with frequency of the various efficiency degradation factors. The attached table shows this variation, combining in the best aluminum all the frequency independent factors as well. These are listed in detail in the rectenna optimization section.

A cursory analysis was also made of the difference of integrated attenuation factors for 2 different sites within U.S., Seattle and El Paso. With unverified assumptions of how often an SPS beam would actually be in a cloud or rain path on a cloudy and rainy day, it was surprising to see the relatively small difference between the 2 sites; even at 5.8 GHz it was only 1.7%. These calculations require further refinement.



D180-24071-3

## Frequency Variation of End-End Efficiency Factors

SPS-1806

BOEING

| FREQUENCY | KLYSTRON<br>EFFICIENCY<br>$\eta_K$ | ATMOSPHERIC<br>LOSS FACTOR<br>$\eta_A$ | RF-DC<br>RECTIFICATION<br>FACTOR R | ANTENNA<br>BEAM<br>EFFICIENCY<br>$\eta_B$ | AVERAGED<br>RECTENNA<br>$\eta_R$ | P <sub>DC</sub> OUTPUT<br>TO GRID |
|-----------|------------------------------------|----------------------------------------|------------------------------------|-------------------------------------------|----------------------------------|-----------------------------------|
| 2.0       | 85.5%                              | 0.985                                  | 100%                               | 94.7%                                     | 0.8025                           | 0.6187 P <sub>p</sub>             |
| 2.45      | 85.0%                              | 0.98                                   | 100%                               | 95.0                                      | 0.802                            | 0.6135 P <sub>p</sub>             |
| 5.8       | 82.4%                              | 0.97                                   | 97.5%                              | 96.1                                      | 0.801                            | 0.5800 P <sub>p</sub>             |

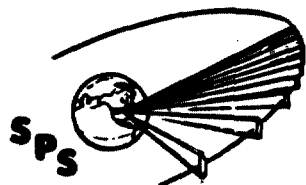
### **MPTS ACQUISITION COST**

The parametric analysis is based on the assumption that about 5 GW of useful power output is desired, that ionospheric heating effects scale as  $1/f^2$ , that a 10 dB taper will provide adequately low sidelobes, and that the thermal dissipation at the spacetenna center will remain the same. This means that the spacetenna remains at a 1 km diameter and the rectenna area is reduced by  $1/f^2$ , and beamwidth by  $1/f$ . The number of subarrays for equivalent dimensional errors must scale with  $f^2$ . The rf power level, to first order, remains constant.

The spacetenna costs, based on modified NASA-JSC numbers, with \$60 kg transportation cost, are derived as in the attached table.

The rectenna costs are based on a fixed number of dipoles per wavelength and the optimized rectenna area derived previously. Dipole and diode costs are taken @ 3¢ independent of frequency.

The results indicate that, although total acquisition cost is minimum at just below 4 GHz, the net return based on the stated assumptions optimizes between 2 and 2.5 GHz. The main value of the more intense ( $130 \text{ mw/cm}^2$  on axis density) 5.8 GHz design is a reduced rectenna size, at an increased technical risk, and greater environmental impact.



D180-24071-3

# MPTS Acquisition Cost Parameters

SPS-1001

BOEING

## • RECTENNA COST AND DIMENSIONAL PARAMETERS

| FREQUENCY<br>GHz | DIPOLES<br>PER M <sup>2</sup> | OPTIMIZED<br>RECTENNA<br>AREA x 10 <sup>6</sup> | % OF<br>AREA TO<br>1ST NULL | DIPOLE AND<br>DIODE COST<br>@ 0.03, \$ M | LAND AND<br>SUPPORT<br>@ \$11.07<br>PER M <sup>2</sup> | TOTAL<br>COST<br>\$ B | COST PER<br>M <sup>2</sup> , \$ | RECTENNA<br>RADIUS, km |
|------------------|-------------------------------|-------------------------------------------------|-----------------------------|------------------------------------------|--------------------------------------------------------|-----------------------|---------------------------------|------------------------|
| 2.0              | 124                           | 114                                             | 60                          | 424                                      | 1,334 M                                                | 1.758                 | 14.74                           | 5.534                  |
| 2.45             | 185                           | 80                                              | 62                          | 444                                      | 938                                                    | 1.380                 | 16.62                           | 5.082                  |
| 5.8              | 1,036                         | 16.2                                            | 69                          | 503                                      | 190                                                    | 0.693                 | 42.15                           | 2.271                  |

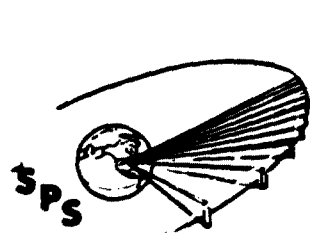
## • SPACETENNA COST FACTORS

| FREQUENCY<br>GHz | KLYSTRON<br>COST<br>\$ M | ARRAY COST<br>M <sup>2</sup><br>(\$) | KLYSTRON<br>POWER kW | KLYSTRON<br>EFFICIENCY<br>(%) | NO. OF<br>SUB-<br>ARRAYS | SPACETENNA<br>COST \$ B |
|------------------|--------------------------|--------------------------------------|----------------------|-------------------------------|--------------------------|-------------------------|
| 2.0              | 530                      | 1303                                 | 100                  | 85.5                          | 4,440                    | 1.245                   |
| 2.45             | 525                      | 1506                                 | 70                   | 85                            | 6,938                    | 1.312                   |
| 5.8              | 515                      | 2078                                 | 35                   | 81.5-82.5                     | 38,883                   | 2.174                   |

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### RETRODIRECTIVE PHASE CONTROL SYSTEM

The basic central phasing system shown coherently receives the uplink and derives from it a signal to drive the transmitters in the central subarray ( $A_0$ ) shown at left. This same phase reference signal ( $\Phi_0$ ) is used to conjugate (phase for retrodirection) the signal which has been received at the  $K^{\text{th}}$  subarray and sent down the connecting transmission line. The conjugated signal returns over the same line resulting in cancellation of line length effects. By the use of an N-node reference proliferation system, it is possible to avoid having to run all the transmission lines back to the central array. It is necessary however to regenerate the phase reference  $\Phi_0$ . A number of different circuits have been identified that can perform the receiving, conjugating, and phase reference regeneration functions. The preferred concept is to use 2 pilot frequencies separated by 200 MHz which avoids phase ambiguity. Three pilot antennas are proposed to permit the shifting of the apparent pilot phase center or position and hence compensate for beam motion due to propagation effects.

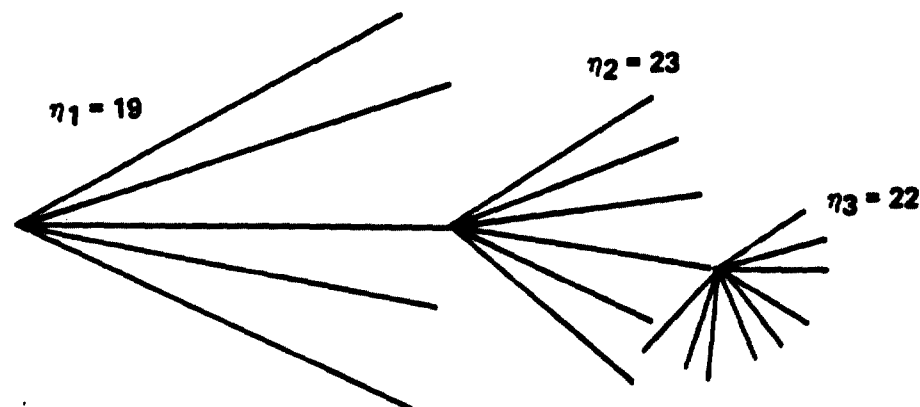


SPS-1865

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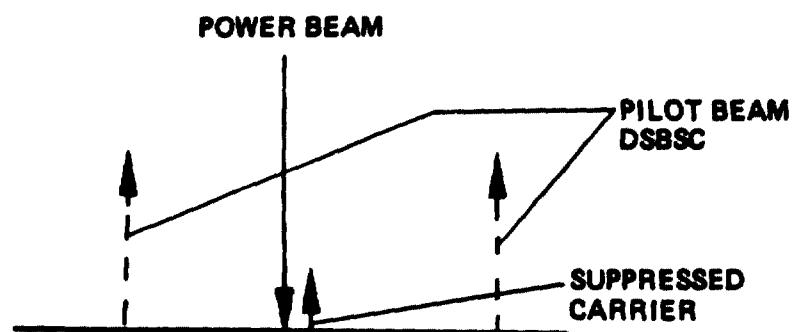
# Features of Baseline Retrodirective Phase Control

BOEING

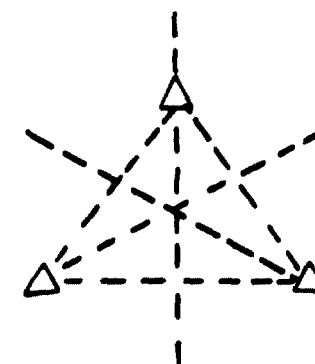


9614 SUBARRAYS

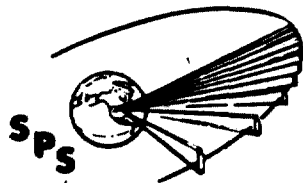
THREE NODE PHASE DISTRIBUTION SYSTEM FOR ERROR CONTROL



DOUBLE SIDEBAND SUPPRESSED CARRIER  
AM MODULATED PILOT BEAM



3-PILOT ANTENNA TRANSMITTER

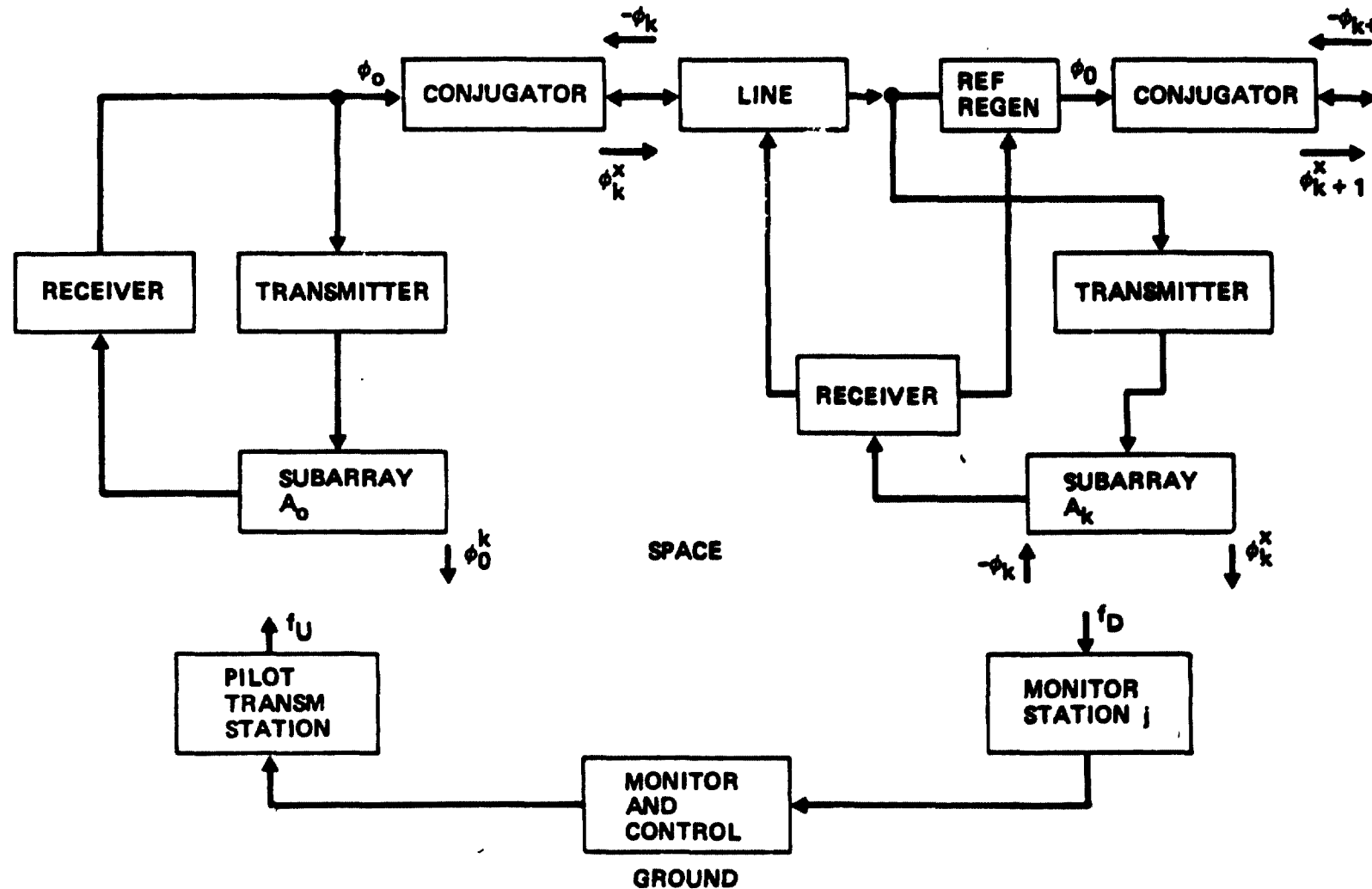


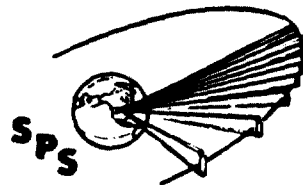
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# Retrodirective Phase Control System

SPS-1874

BOEING





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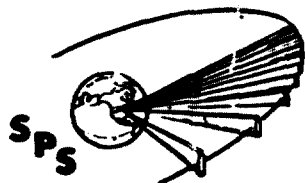
## SPS Phase System Comparisons

SPS-1966

**BOEING**

|                 | PILOT BEAM                 | PHASE<br>CONJUGATION                                | REF.<br>DISTRIBUTION                | FEATURES                                         |
|-----------------|----------------------------|-----------------------------------------------------|-------------------------------------|--------------------------------------------------|
| JPL             | DUAL<br>DSBSC              | EXACT $\odot$ IF<br>DIFFERENCE<br>FREQ.             | SERIES TREE<br>CENTRAL<br>PHASING   | PLL<br>MINIMUM<br>CABLE                          |
| LIN COM         | SINGLE                     | APPROX. $\odot$ RF<br>CORRECTED BY<br>PHASE SHIFTER | SERIES STRING<br>CENTRAL<br>PHASING | MONOPULSE<br>COMPUTER                            |
| BOEING/<br>G.E. | DUAL<br>DSBSC<br>(3-PILOT) | EXACT $\odot$ IF<br>DIFFERENCE<br>FREQ.             | 3 LAYER TREE<br>CENTRAL<br>PHASING  | SYSTEMATIC<br>PROPAGATION<br>ERRORS<br>TAKEN OUT |





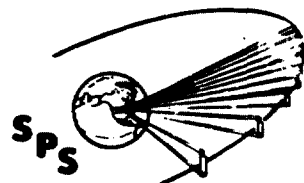
SPS-1876

D180-24071-3

## Phase Control Studies

**BOEING**

- **ALTERNATE APPROACHES**
  - **RETRODIRECTIVE SYSTEM**
  - **GROUND COMMANDED SYSTEM**
  
- **ERROR BUDGET ESTIMATION**
  - **COMPONENT EFFECTS**
  - **PROPAGATION MEDIUM**
  - **ERROR MINIMIZATION**



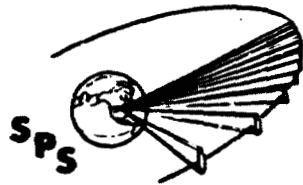
D180-24071-3

## Atmospheric Attenuation Factors

SPS-1888

**BOEING**

|                       | GHz  | SEATTLE | EL PASO |
|-----------------------|------|---------|---------|
| INTEGRATED            | 2.45 | 0.985   | 0.985   |
| ATMOSPHERIC<br>FACTOR | 5.8  | 0.959   | 0.976   |

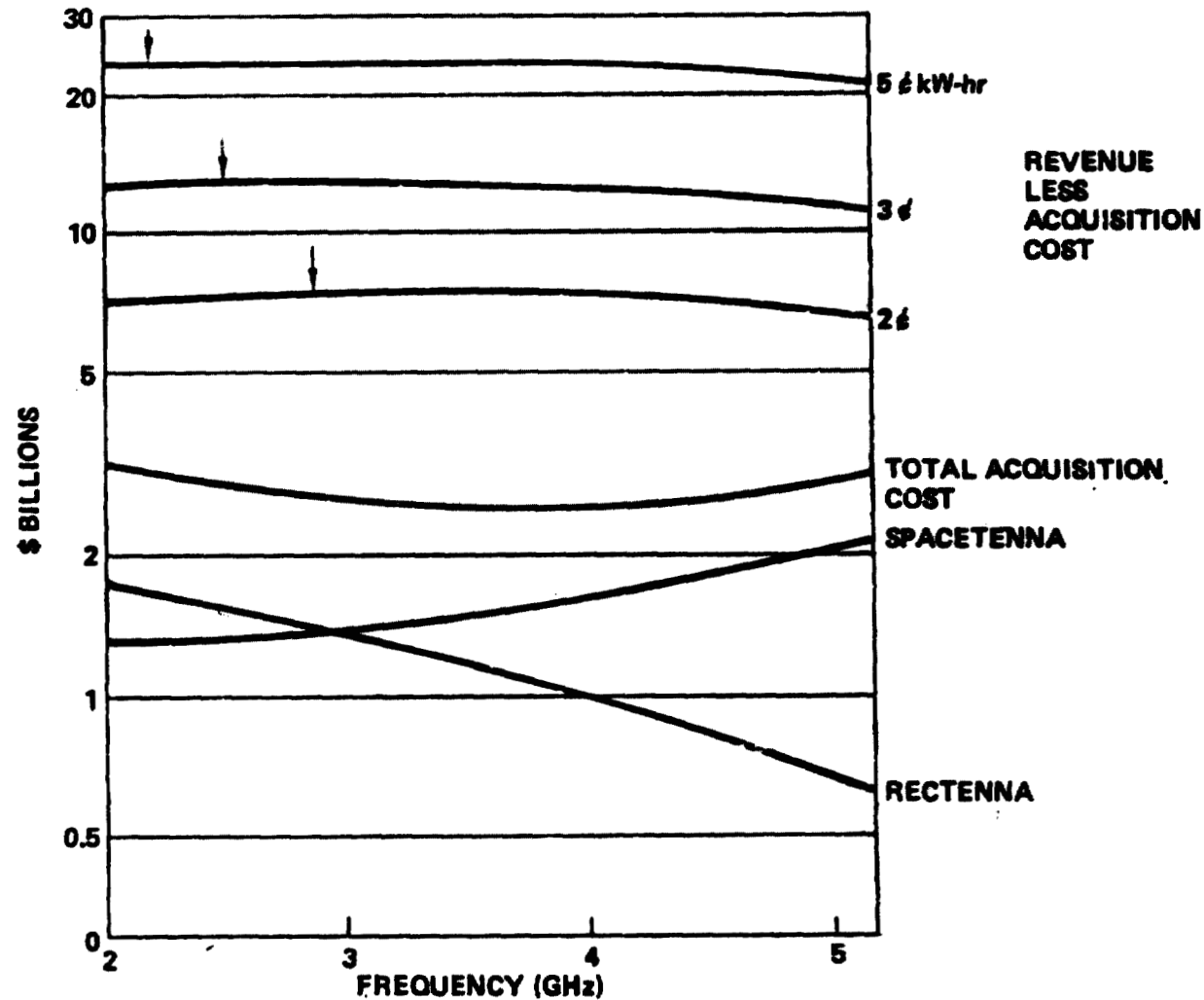


SPS-1892

D180-24071-3

# Acquisition Cost and Revenue Variation With Frequency

BOEING



5 GW NOMINAL DC 1 KM SPACENNA 10 dB GAUSSIAN TAPER

### PHASE ERROR BUILDUP IN THREE LAYER REFERENCE DISTRIBUTION SYSTEM

The proposed phase control system utilizes three layers as illustrated. 95% of the subarrays are at the third level. Errors due to conjugators, transmission line reflections, diplexer and transmitter add in the distribution system as indicated by the RSS formula. Using the following estimated values of component phase errors

Conjugator ( $g_c = 0.6^\circ$ )

Transmission Line ( $g_l = 2.5^\circ$ )

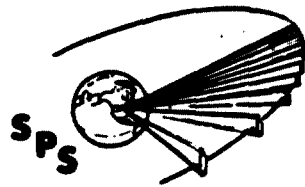
Diplexer ( $g_d = 1.8^\circ$ )

Transmitter ( $g_p = 1.6^\circ$ )

The cumulative RSS total is  $7.0^\circ$

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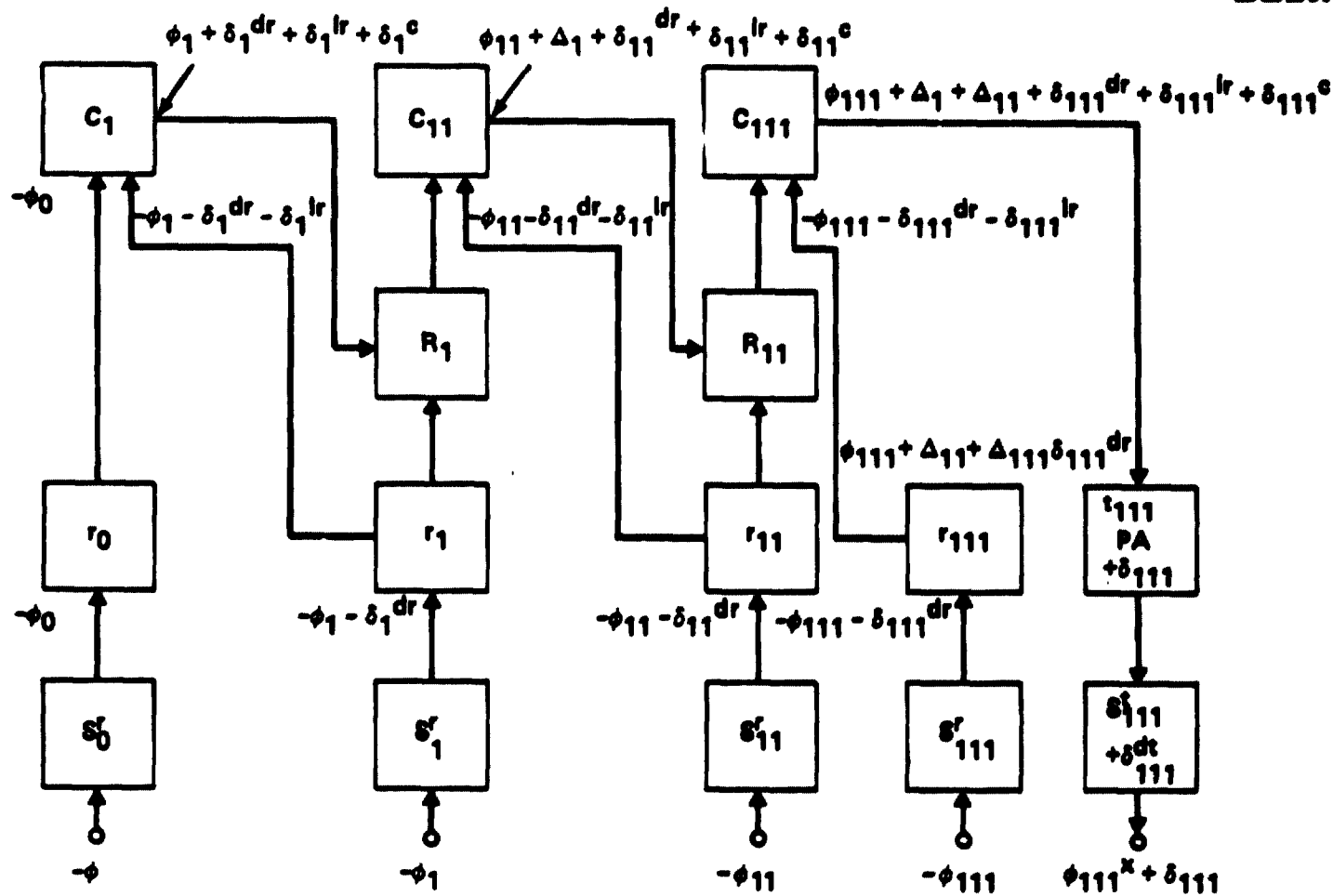


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# Phase Error Buildup in Three Layer Reference Distribution System

SPS-1871

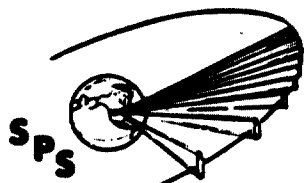
BOEING



$$\text{RSS ERROR} = \sqrt{3\delta_0^2 + 6\delta_1^2 + 2\delta_d^2 + \delta_p^2}$$

**REDUCTION OF POINTING ERRORS BY THREE PILOT STATION SYSTEM**

The pointing errors listed are not correctable by the phase control system. They can produce power beam displacement from the rectenna. A ground system of 3 pilot stations is proposed to correct for these beam position errors should they develop to be of sufficient magnitude. This system utilizes monitor antennas placed about the rectenna. Displacement of the beam results in a corrective counter displacement of the pilot phase center (or effective position). The latter is accomplished by relative amplitude and phase control of the radiation from the 3 separated pilot antennas.



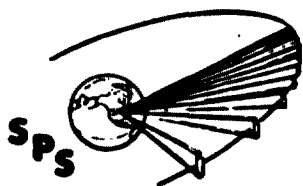
D180-24071-3

# Reduction of Pointing Errors by Three Pilot Station System

SPS-1873

BOEING

| POINTING ERRORS (deg)                                                |                       |                        |
|----------------------------------------------------------------------|-----------------------|------------------------|
| SOURCE                                                               | 1 PILOT STATION       | 3 PILOT STATION        |
| DOPPLER ( $i = 2.2$ deg, $r_m^i = 13.6$ m/s, $2 f_{Dop} = 112$ Hz)   | $1.43 \times 10^{-6}$ | $7.15 \times 10^{-8}$  |
| ABERRATION ( $Z_m^i = 100$ m/s)                                      | $19.3 \times 10^{-6}$ | $9.65 \times 10^{-8}$  |
| IONOSPHERIC DIFFERENTIAL (0.1 deg 1 WAY REFRACTION)                  | $2.35 \times 10^{-3}$ | $1.17 \times 10^{-4}$  |
| ATMOSPHERIC DIFFERENTIAL (0.3 deg 1 WAY REFRACTION, 2% IRREGULARITY) | $6.00 \times 10^{-3}$ | $3.00 \times 10^{-4}$  |
| POINTING ERROR<br>(deg)                                              | PEAK                  | $8.35 \times 10^{-3}$  |
|                                                                      | RSS                   | $4.175 \times 10^{-4}$ |
|                                                                      |                       | $3.221 \times 10^{-4}$ |



D180-24071-3

## Random Errors for 3 Layers Phase Distribution

SPS-1872

BOEING

| PHASE ERRORS (deg)                       |                      |
|------------------------------------------|----------------------|
| SOURCE                                   | DEG                  |
| PHASE JITTER                             | 1.13                 |
| TRANSMITTER NOISE ( $\sigma/n = 30$ dB)  | 0.36                 |
| CONJUGATORS ( $\delta_c = 0.6$ deg)      | 1.04                 |
| LINES ( $\delta_l = 2.54$ deg)           | 6.22                 |
| DIPLEXERS ( $\delta_d = 1.81$ deg)       | 2.56                 |
| TRANSMITTERS ( $\delta_p = 1.6$ deg)     | 1.60                 |
| DIFFERENTIAL DOPPLER ( $V_d = 6.25$ m/s) | 0.18                 |
|                                          | PEAK:13.09 RMS: 7.09 |

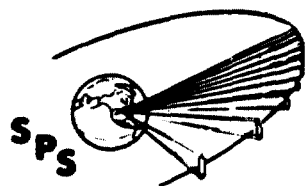
PHASE ERROR CAUSED LOSS: 1.53%



### GROUND COMMANDED PHASE CONTROL SYSTEM

It is possible to transfer much of the phase control system complexity to the ground. The ground commanded alternate system shown here utilizes a unique identifying tone that is modulated on the output of each subarray. From this low level signal, the relative phase of a subarray's contribution to the total field is derived. Phase shifter adjustments of the radiated carrier phase are then commanded from the ground. To hold the number of channels required to a reasonable value a 100 slot time division multiplex scheme is proposed. With under 100 tones, the sampling period of this system would be  $\approx 25$  sec unless an interlaced technique were used which could possibly reduce it to 0.25 sec. Convergence of this type of phase control system needs to be studied with an appropriate structural dynamic model.

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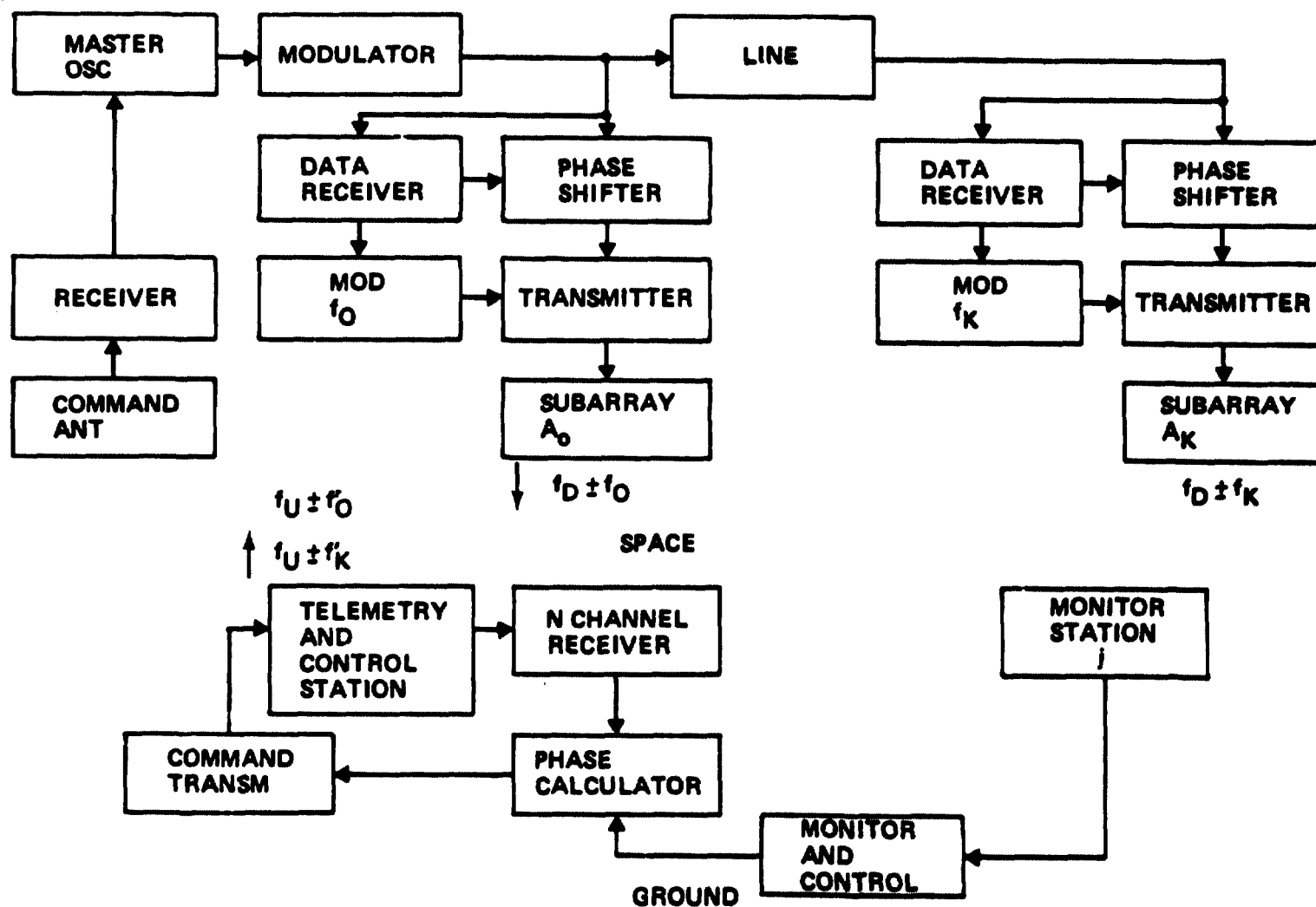


D180-24071-3

# Ground Commanded Phase Control System

SPS-1876

BOEING



### PROJECTED CONVERSION EFFICIENCY OF OPTIMIZED RECTENNA ELEMENT

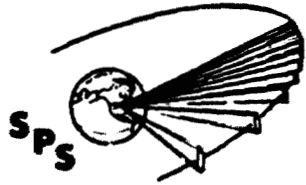
The data for modeling the rf-dc conversion efficiency is based on experimental work by Raytheon and JPL with projections of improved diode and circuit performance made by NASA-JSC. The resulting equation is:

$$\eta = 91 - 4.1e^{-415 P_d}, \text{ and}$$
$$\eta = 0 \text{ for } P_d < 10^{-3}, \text{ where}$$

$\eta$  is in percent and  $P_d$  is incident power density in  $\text{mW}/\text{cm}^2$ .

The conversion efficiency equation was incorporated into the modified "Bigmain" program in order to calculate the average rectenna conversion efficiency as a function of rectenna area.

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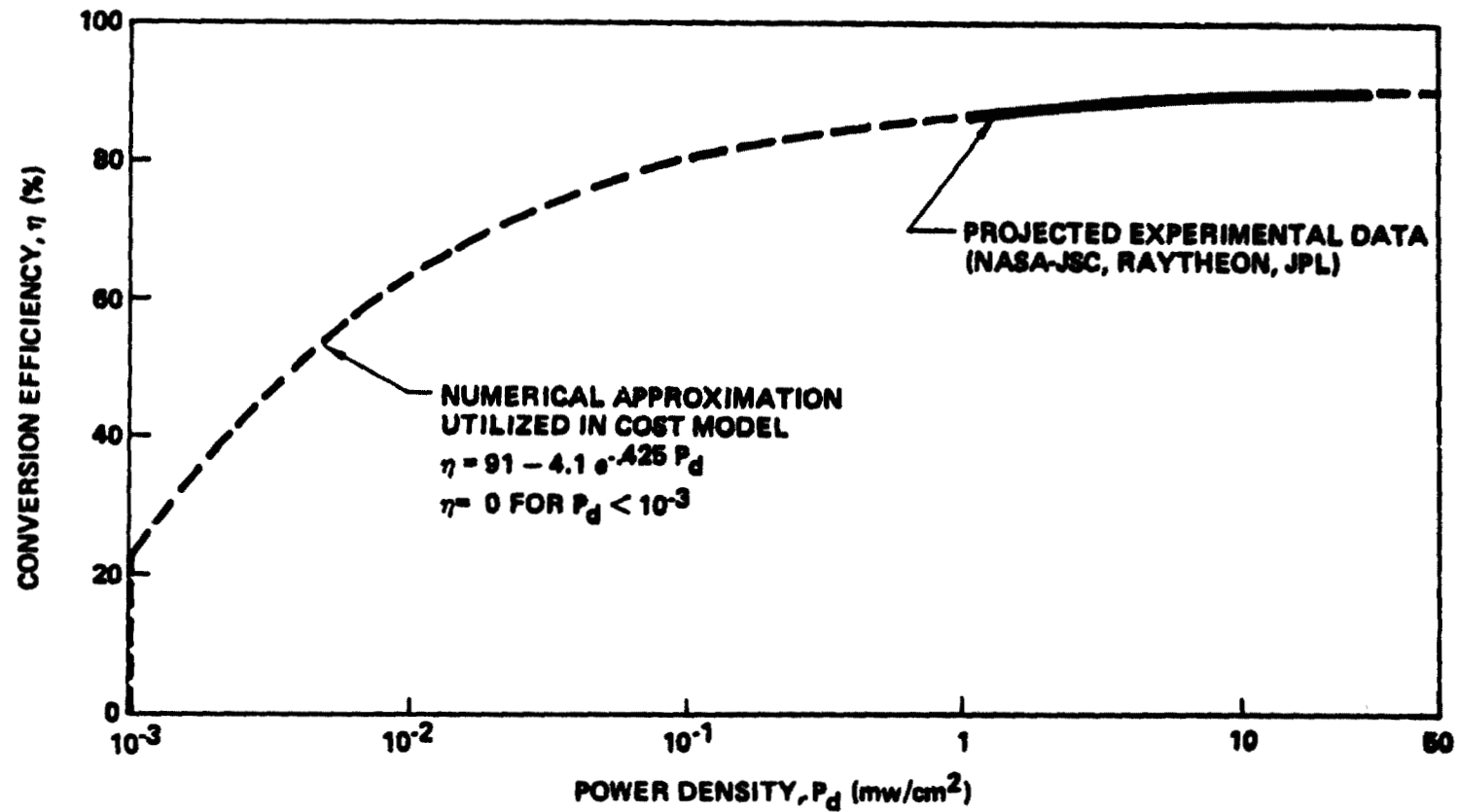


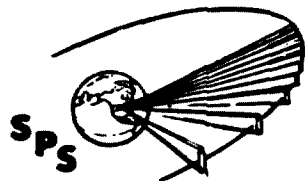
SPS-1893

D180-24071-3

## Projected Conversion Efficiency of Optimized Rectenna Element

BOEING





SPS-1877

D180-24071-3

## Rectenna Analysis Tasks

**BOEING**

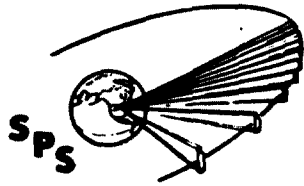
- IMPROVED RECTENNA EFFICIENCY MODEL
- RECTENNA SIZE OPTIMIZATION
- EDGE DIFFRACTION EFFECTS
- HARMONIC GENERATION
- MECHANICAL DESIGN
- ALTERNATE APPROACHES

**PROJECTED AVERAGE RECTENNA CONVERSION EFFICIENCY  
AS A FUNCTION OF RECTENNA AREA AND ON AXIS POWER DENSITY**

The family of curves shows the average conversion efficiency as a function of rectenna area for the five different values of on axis power densities and their associated spacetenna radiated power. For the baseline 10 dB gaussian taper, the spacetenna beam efficiency has a one-to-one correspondence to the rectenna area as is indicated by the top scale. The averaging process, carried over 50 nominal ring sections of the rectenna, derives an average conversion efficiency

$$\eta_R = \frac{\sum_{o}^R \eta P_d A}{\sum_{o}^R P_d A} \quad \text{for any desired rectenna radius } R.$$

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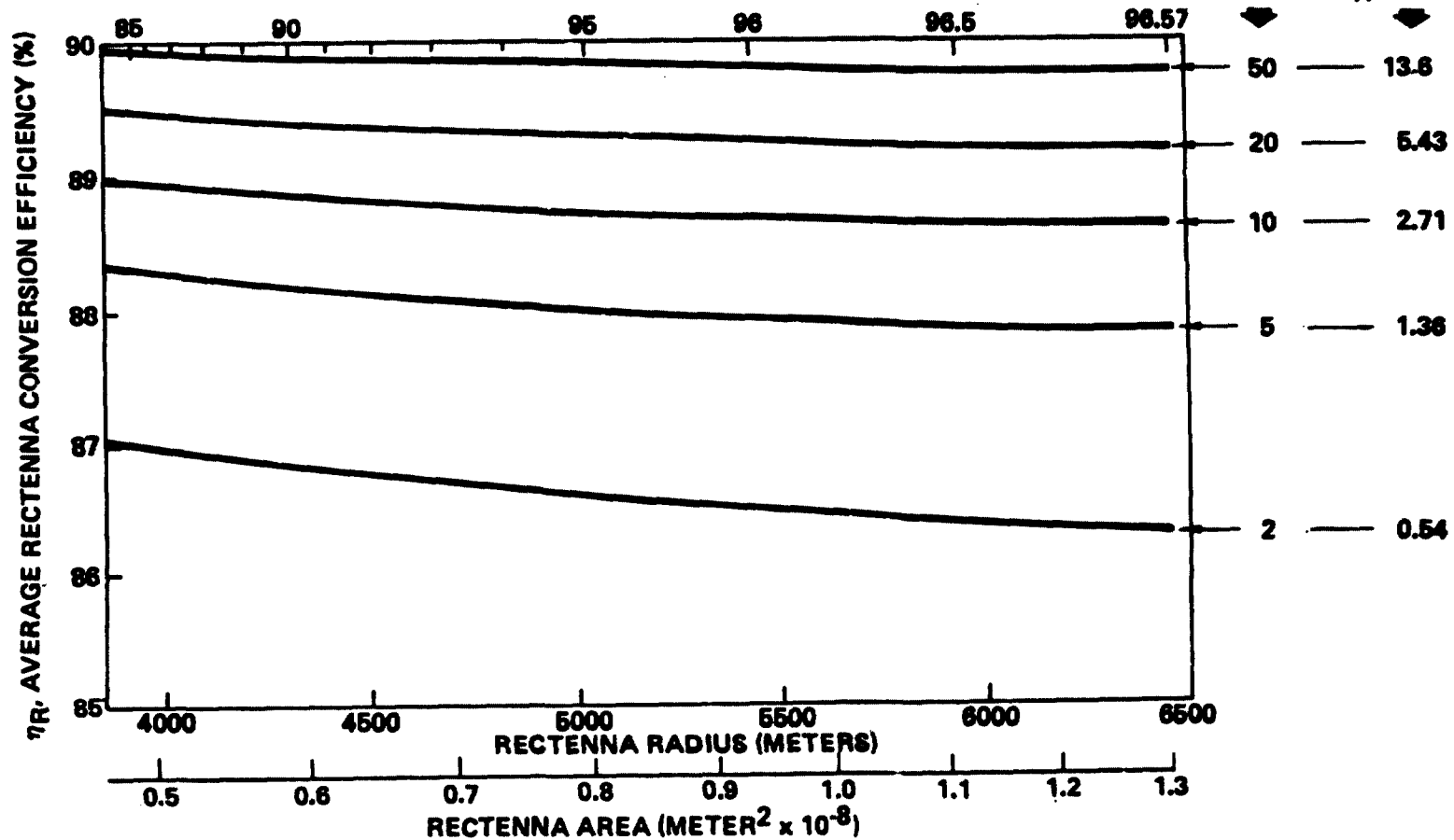
SPS-1894

# Projected Average Rectenna Conversion Efficiency Versus Rectenna Area

BOEING

$\eta_B$  = SPACETENNA BEAM EFFICIENCY (%)  
(10 dB GAUSSIAN TAPER, 1 km DIAMETER,  $f = 2.45$ )

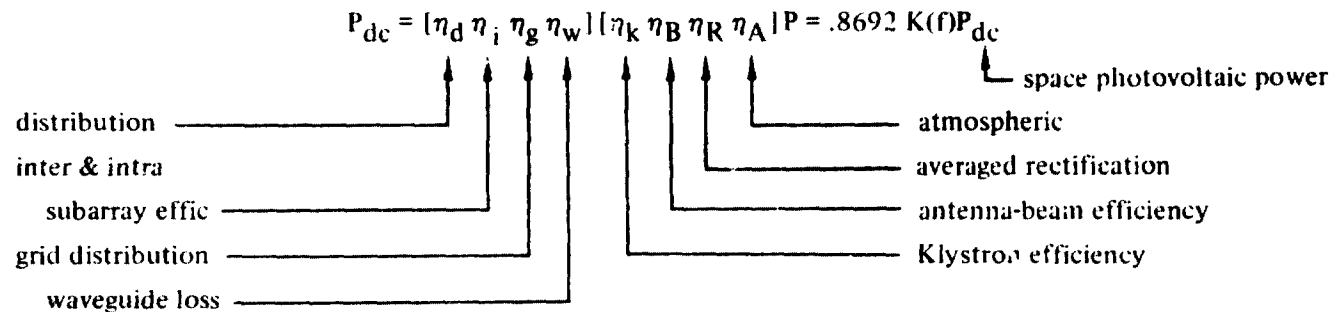
ON AXIS POWER DENSITY  
(mw/cm<sup>2</sup>)  $P_{rf}$  GW



## RECTENNA OPTIMIZATION

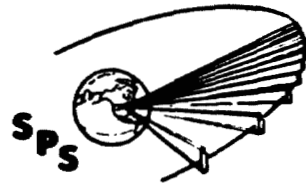
The optimum rectenna size is selected on the basis of matching the present value of 30 year revenue per spare meter with the acquisition cost per square meter. This process is shown for 2.45 GHz and 5.8 GHz, for various values of revenue. For the selected parameters, the rectenna area is only 62% of the area out to the first null.

The values of the various efficiency degradation factors contributing to the net dc power to the ground network are treated parametrically with frequency as follows:



The frequency varying portion is given in the attached table, from which the useful dc power output can be plotted as a function of rectenna radius. From this plot, the revenue per  $m^2$  can be obtained and equated to the rectenna cost per square meter.

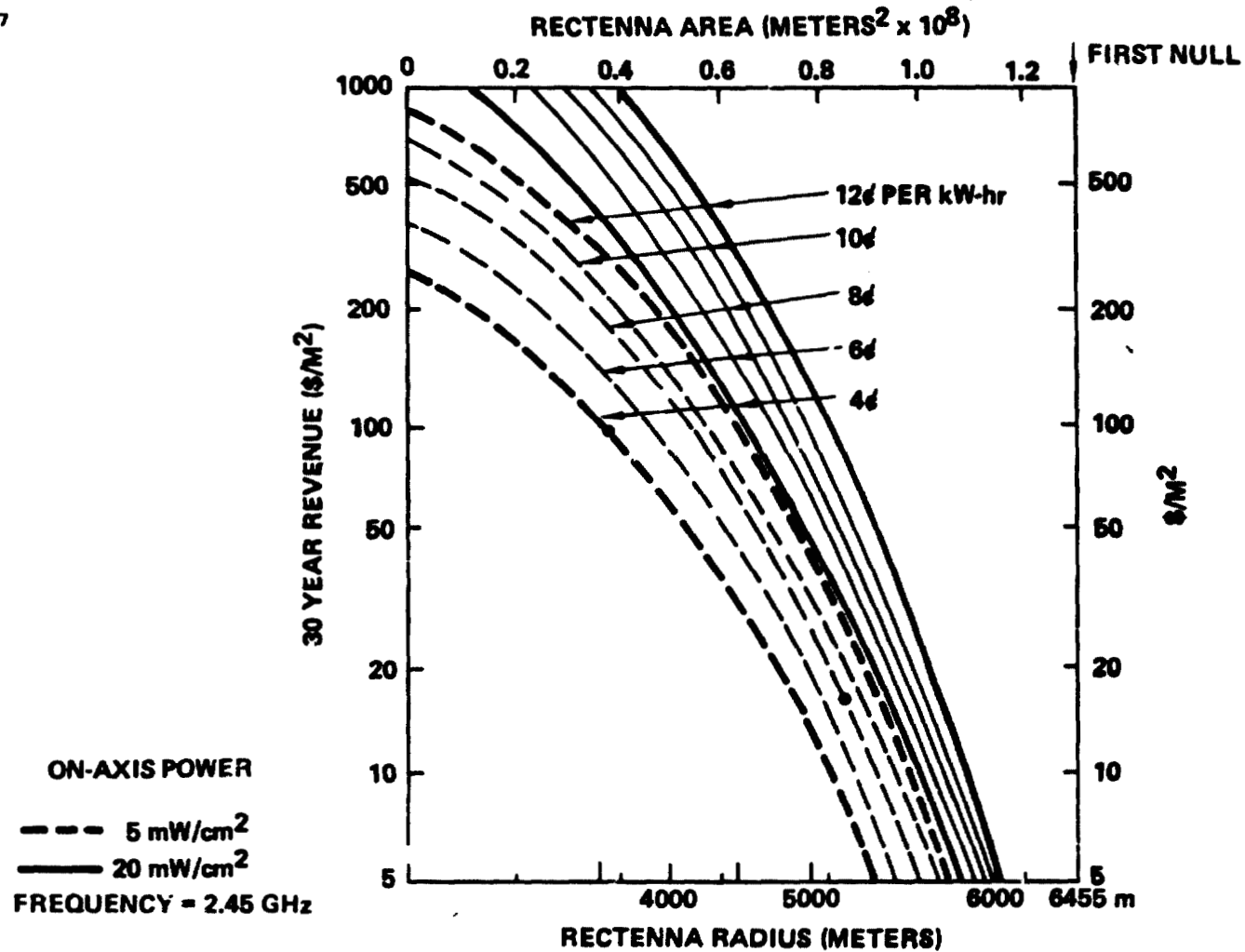


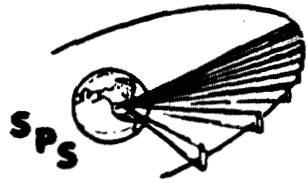


## Optimum Rectenna Dimension for Given Rectenna Cost

SPS-1897

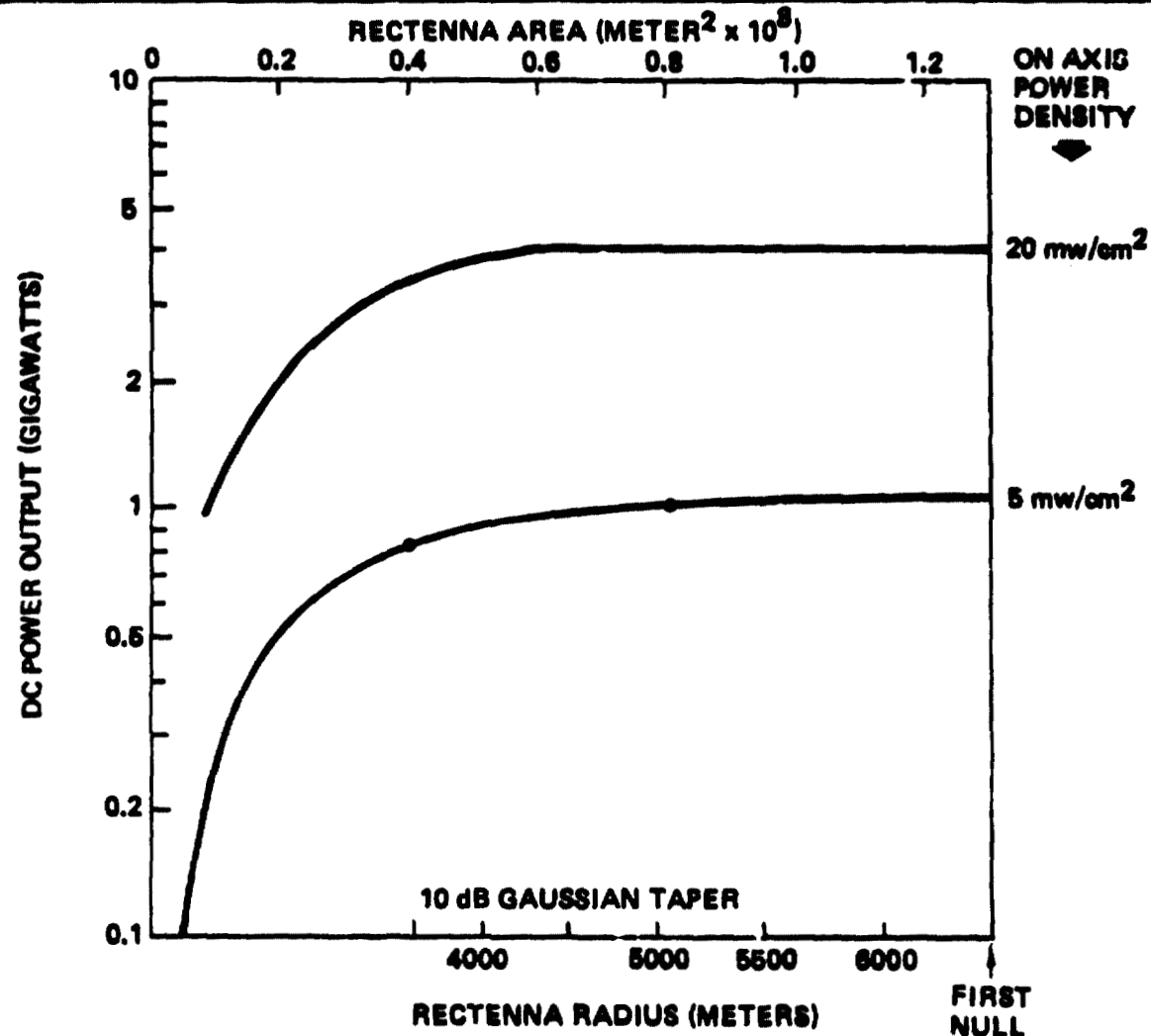
BOEING





SPS-1896

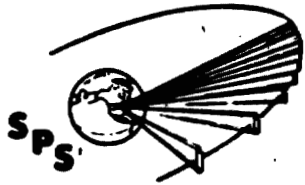
# Power Output Versus Rectenna Size



### HOGLINE RECTENNA CONFIGURATION

A suggested hogline rectenna configuration is shown. An offset cylindrical parabola is fed by a trough or line horn. A dipole/diode panel of 12 rows is located in this example where the incident power density has been increased by a factor of 8.7. Spillage, blockage and taper losses can all be made negligible with this configuration. The greater power density permits higher rectification efficiency, or lower power demonstrations at the same efficiency. The dipole and diodes are protected from weather and EMP and there are significantly fewer of them. Size is a compromise between the degree of concentration desired, edge diffraction losses and the latitude stability of the SPS satellite.

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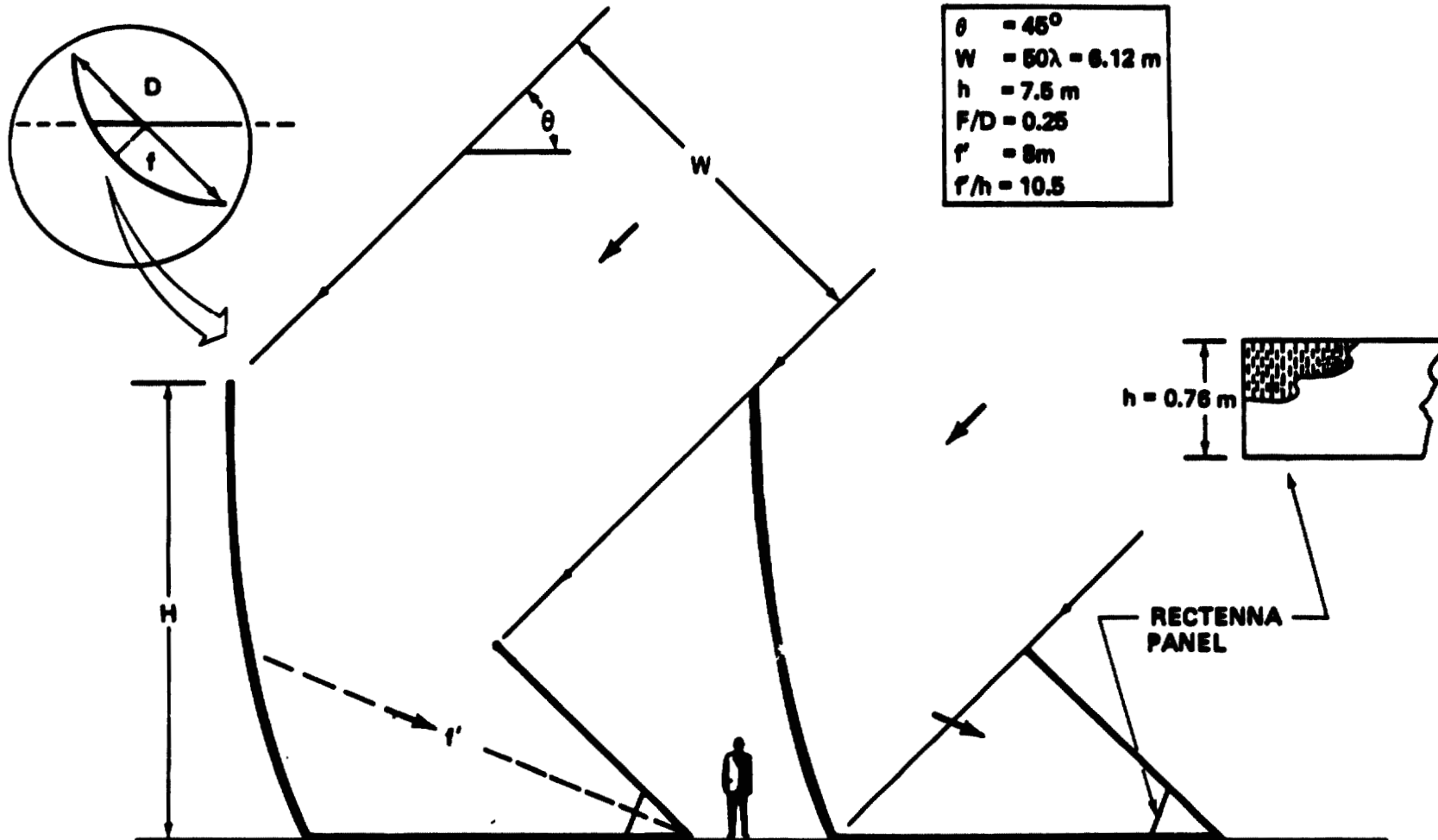


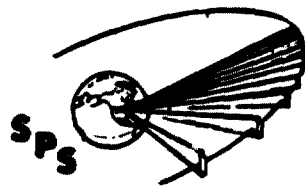
D180-24071-3

# Hogline Rectenna Configuration for SPS

SPS-1800

BOEING



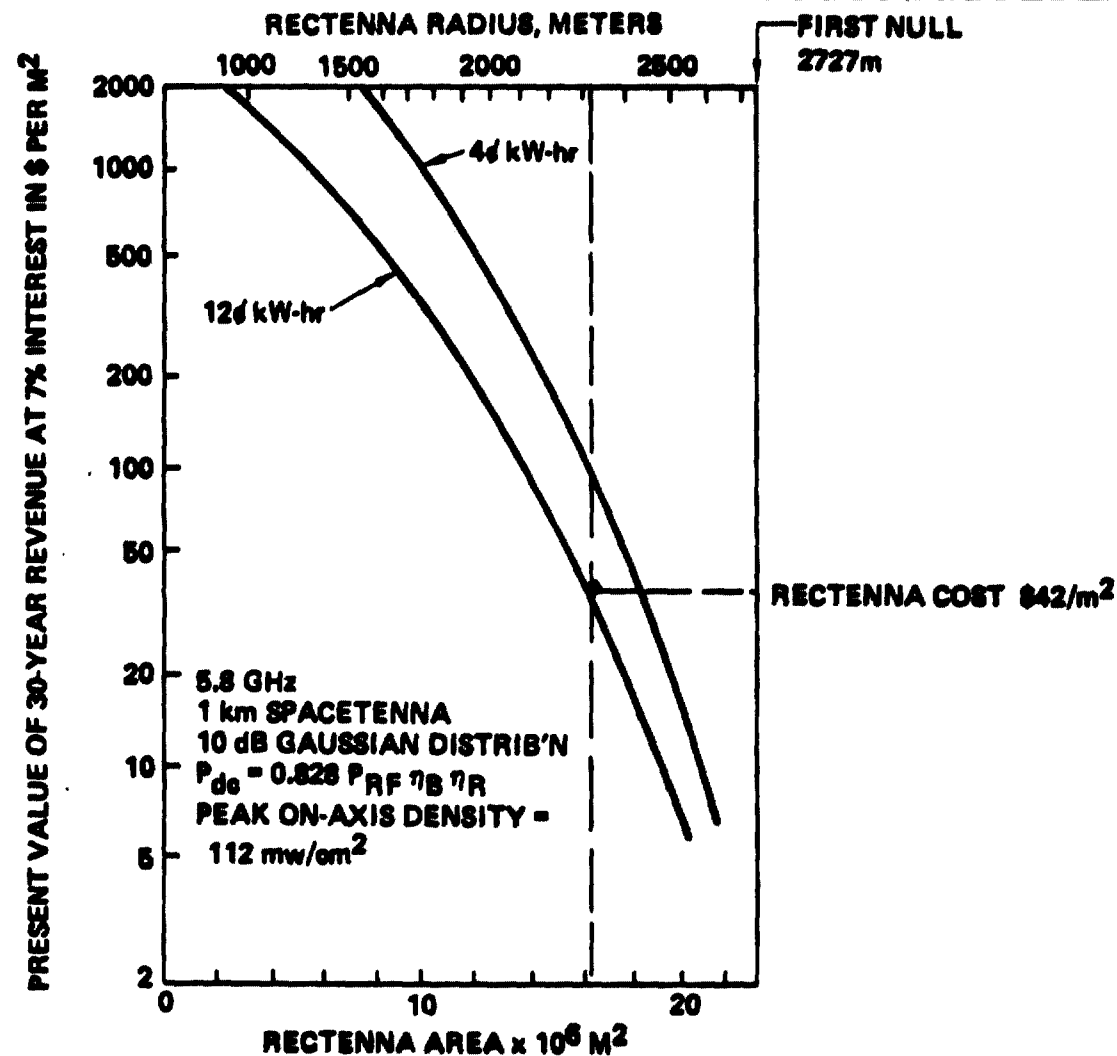


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# Rectenna Size Optimization

SPS-1808

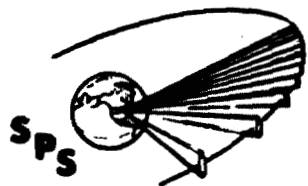
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### **HOGLINE ANTENNA POINTING CONSIDERATIONS**

Due to the vertical plane pattern directivity, variations in satellite latitude will induce pointing losses. These losses are estimated by considering the fraction of the converging beam that misses the diode/dipole panel. The losses are a function of the relative panel size ( $f/h$ ). The geometry shown in the previous figure would require a stability of  $0.05^\circ$  to hold losses to 1.5%.

Looked at from the simple pattern of the  $50\lambda$  aperture loss point of view, it is estimated that an error of  $0.08^\circ$  could be tolerated for a 1.5% power loss.

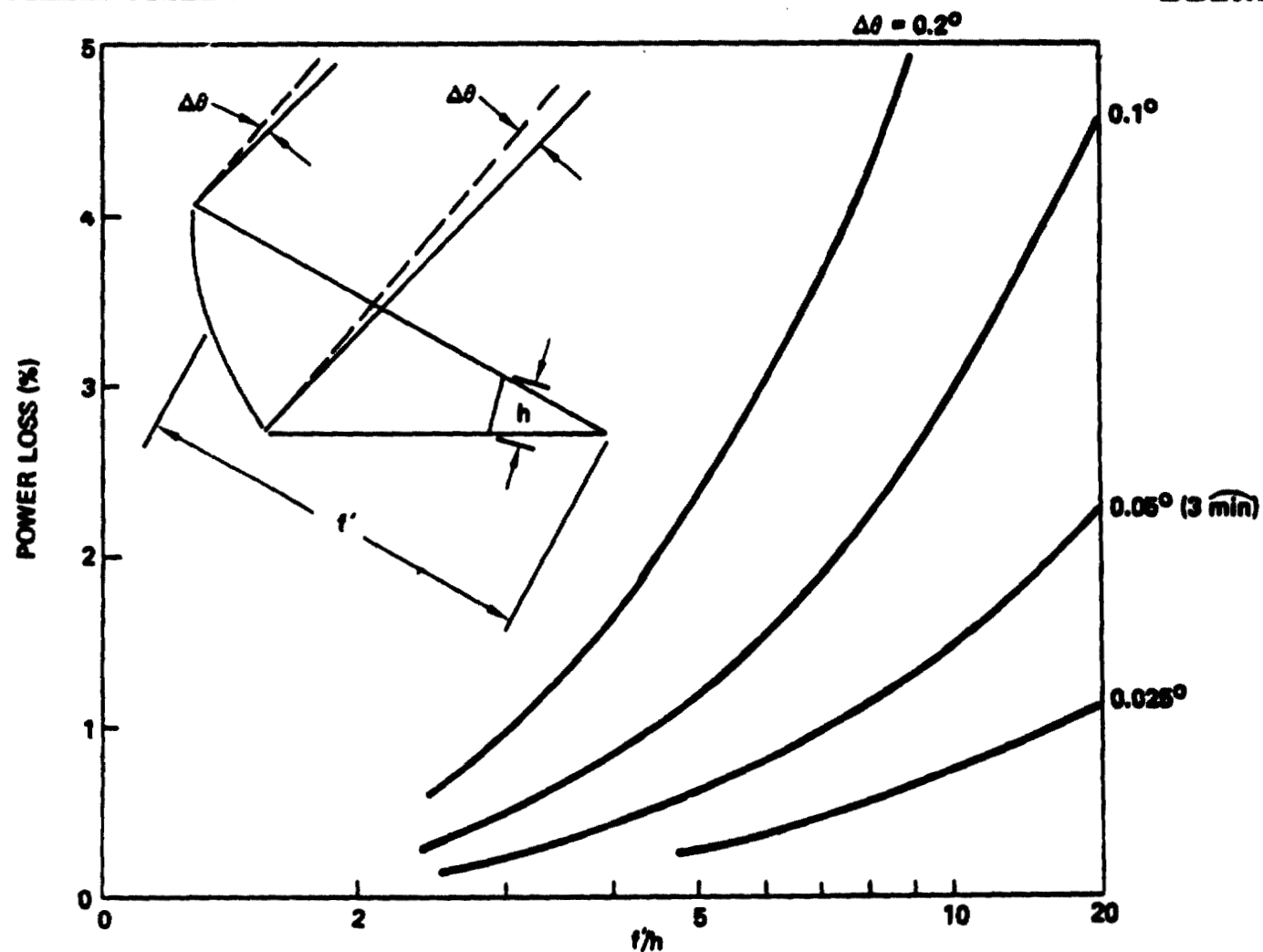


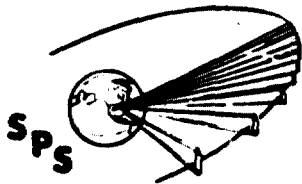
D180-24071-3

# Pointing Power Loss Versus Rectenna Panel Size

SPS-1888

BEING





SPS-1848

D180-24071-3

## MPTS Follow-On Topics

BOEING

### ● ARRAY ANALYSIS

- PHASE CONTROL SYSTEM DEFINITION
- SPS PRECURSOR SPACETENNA CONFIGURATION AND PATTERN ANALYSIS
- GRATING LOBE MAGNITUDE ASSESSMENT
- DYNAMICS OF GROUND CONTROLLED PHASE COMPENSATION
- SPACETENNA BACK RADIATION ANALYSIS
- FAR SIDELobe ROLL-OFF

### ● COMPUTER PROGRAM EXTENSION

- MODEL ARRAY AT KLYSTRON LEVEL
- EFFECT OF SUBARRAY OFFSET AND INTRA-SUBARRAY ERRORS
- SYSTEMATIC AMPLITUDE AND PHASE ERROR EFFECTS
- EXTEND CAPABILITY TO OTHER THAN PRINCIPAL PLANES

### ● RF TRANSMITTER

- KLYSTRON FAILURE MODE ANALYSIS AND PROTECTION
- DETAILED TUBE SPECIFICATION TO MANUFACTURER
- TEST PLAN FOR GROUND AND SPS PRECURSOR TESTS
- HARMONIC SUPPRESSION FEASIBILITY

### ● COMPONENT TEST PROGRAM

- COMPOSITE WAVEGUIDE MICROWAVE POWER TEST
- CROSSGUIDE DESIGN AND ANALYSIS, DIPLEXER DESIGN

### ● RECTENNA COLLECTION EFFICIENCY OPTIMIZATION

- MULTIPLE DIPOLE PER DIODE CONFIGURATION
- HOGLINE ANTENNA
- ELEMENT IMPEDANCE GAIN VERSUS SPACING

### ● SYSTEM STUDIES

- AC GENERATION AND DITHER FEASIBILITY
- TURN ON/OFF PROCEDURES
- FLOPPY REFLECTOR LEO TEST

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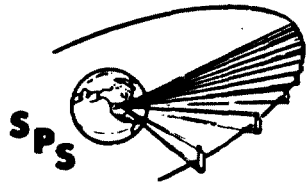


# **Development Requirements for Initial SPS Commercialization**

**D. L. Gregory**

**PAST AND FUTURE ELECTRIC POWER ALTERNATIVES USE PILOT PLANTS**

Today's nuclear power reactors are the culmination of a series of developmental reactors which preceded them. In the ground solar power program the thermal engine "tower top" systems are to be preceded by the five megawatt thermal test facility at Albuquerque, New Mexico and by a ten megawatt (electric) output pilot plant at Barstow, California. These will lead to a commercial unit to be rated at 100 megawatts. Thus the pilot plant is 1/10 of the size of the eventual commercial unit. Ground solar power plants of the tower top type have been investigated as large as 1,000 megawatts. The future of the liquid metal fast breeder reactor is uncertain, however, for that program a series of developmental systems is envisioned. The total fusion power development program is as yet undefined, however, a series of experimental reactors and a demonstration reactor are envisioned.



SPS-1859

D180-24071-3

# Past and Future Electric Power Alternatives Use "Pilot" Plants

BOEING

(1976 DATA)

## PAST: PROGRESSION TO THE COMMERCIAL LIGHT WATER REACTOR

- EXPERIMENTAL REACTORS,
- DEMO REACTOR,
- PROTOTYPE PLANT,

SHIPPINGPORT, PA  
OYSTER CREEK, NJ

## FUTURE: GROUND SOLAR POWER (TOWER TOP TYPE)

- TEST FACILITY
- PILOT PLANT
- COMMERCIAL DEMONSTRATOR

(5 MW<sub>e</sub>) ALBUQUERQUE, NM  
(10 MW<sub>e</sub>) BARSTOW, CA  
(100 MW<sub>e</sub>)

### BREEDER REACTOR (LIQUID METAL FAST BREEDER)

- FAST FLUX TEST FACILITY
- CLINCH RIVER BREEDER
- PROTOTYPE COMMERCIAL BREEDER

400 MW<sub>e</sub>  
380 MW<sub>e</sub>  
1,200 MW<sub>e</sub>

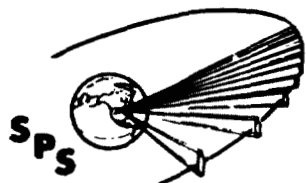
### FUSION (MAGNETIC)

- EXPERIMENTAL POWER REACTOR 1 (20-50 MW<sub>e</sub>)
- EXPERIMENTAL POWER REACTOR 2 (▷ 100 MW<sub>e</sub>)
- DEMONSTRATION REACTOR (▷ 500 MW<sub>e</sub>)

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### **PRECURSOR SOLAR POWER SATELLITE**

Many scenarios can be postulated for the development sequence leading to full size solar power satellites. The most aggressive of these would advance directly from small scale tests with the shuttle to a full size unit in a very rapid development program. At the other extreme is a "long chain" development beginning with shuttle and ground tests, then small scale power modules, then pilot plants, then commercial demonstrators and finally full size power satellites. Some development scenarios for SPS include participation in SPS funding by utility firms. In order to achieve this participation it may be necessary to demonstrate commercial viability of SPS to these utility firms. To do so a precursor satellite may be required. A precursor satellite would be a subscale unit put on line in geosynchronous orbit before the full size satellite. It would operate in the fashion of the full size satellite, that is direct an energy beam to earth to a ground rectenna. The demonstration/operational period might be six months to 2 years and during this time the majority of the major elements of a full sized solar power satellite would be exercised. From this operation, analyses could be performed which would indicate eventual full commercial viability for the SPS. It is also possible to construct scenarios in which the full size satellite is preceded by a precursor which serves only as part of the development leading to the full sized unit. Other scenarios go directly to the full size SPS without the use of the precursor.



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## Precursor Solar Power Satellite

SPS-1851

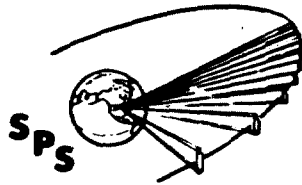
*BOEING*

- MANY SPS DEVELOPMENT SCENARIOS CAN BE IDENTIFIED.
- SOME OF THESE INCLUDE FUNDING PARTICIPATION BY UTILITY FIRMS.
- FOR THESE SCENARIOS, A PRECURSOR SPS COULD BE USED TO ESTABLISH EVENTUAL SPS COMMERCIAL VIABILITY.
- IN OTHER SCENARIOS, A PRECURSOR SPS WOULD SERVE PRIMARILY AS A DEVELOPMENT ITEM.
- SOME SPS SCENARIOS WOULD NOT INVOLVE A PRECURSOR.

### PRECURSOR SPS LIMITATIONS

Due to its small size and the scaling effects related to microwave transmission a precursor or subscale satellite cannot develop more than perhaps 1 or 2 milliwatts per square centimeter in its central beam. This is well below the quantity probably required to excite ionospheric heating. Also because of its small size and the associated development costs it is probable that a precursor unit cannot be procured for less than approximately \$100,000 per milowatt; as a consequence the electric energy produced during the short term of operation would be quite expensive.

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SPS-1831

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## Precursor SPS Limitations

BOEING

### THE PRECURSOR SPS CANNOT:

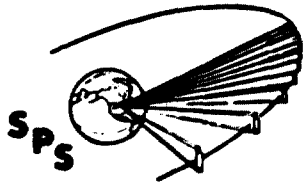
- PRODUCT CENTRAL BEAM STRENGTHS WHICH EXCITE IONOSPHERIC HEATING
- PRODUCE "ECONOMIC" ELECTRIC ENERGY (FORTUNATE TO OBTAIN \$100,000/kW)

**STUDY ASSUMPTION: PRECURSOR SPS**

For this study we have assumed that a demonstration of eventual commercial viability will be the goal of the precursor solar power satellite. In addition to a demonstration of eventual commercial viability the operation of a subscale unit will accomplish the secondary goals identified here.

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SPS-1828

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## Study Assumptions: Precursor SPS

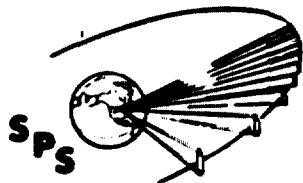
BOEING

- A DEMONSTRATION OF EVENTUAL COMMERCIAL VIABILITY IS THE PRIMARY GOAL OF A PRECURSOR SPS (PSPS)
- SUCH A DEMONSTRATION WILL ACCOMPLISH SECONDARY GOALS:
  - SHOW THAT THE SPS CONCEPT "WORKS"
  - VERIFY DESIGN ASSUMPTIONS
  - REFINE DATA ON ENVIRONMENTAL EFFECTS ON SPS
    - IONOSPHERIC EFFECTS ON BEAM
    - PLASMA PHENOMENA
    - SPACECRAFT CHARGING
  - ESTABLISH OPERATIONAL EXPERIENCE
    - HIGH LAUNCH RATES
    - ON-ORBIT CONSTRUCTION (HIGH "THROUGHPUT")
    - OPERATION AND MAINTENANCE
    - COLLISION AVOIDANCE

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**COMMERCIAL VIABILITY OF SPS**

In order to develop the concept for a precursor solar power satellite which will demonstrate eventual commercial viability for an SPS program it is necessary to first define commercial viability. The definition here is literally that the product of the SPS be in sufficient demand. In order to be in sufficient demand its price, expressed as dollars per kilowatt hours, must be equal or below some criterion. Generally for SPS this criterion may be viewed as approximately 10 cent per kilowatt hour. However, our studies have identified satellite programs wherein the cost is probably no more than 5 cent per kilowatt hour. The price per kilowatt hour of energy produced by any power plant is made up of the annual cost for that plant divided by the kilowatt hours produced by that plant in a year. Hence, good availability, that is the capability of producing a large fraction of the total plant capacity in kilowatt hours per year, is as important as the cost amortization per year.



SPS-1829

D180-24071-3

## Commercial Viability of SPS

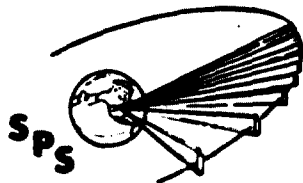
BOEING

- **COMMERCIAL VIABILITY: A COMMODITY, IN DEMAND AT OR BELOW A GIVEN PRICE, IS PRODUCIBLE AT OR BELOW THAT PRICE**
- **COMMODITY PRICE FOR ELECTRICAL ENERGY: EXPRESSED AS DOLLARS PER KILOWATT HOUR OF DELIVERED ENERGY = \$/kWh**
- **PRICE = COST, PLUS FACTORS SUCH AS PROFITS**
- **COSTS FOR SPS ARE PRIMARILY INITIAL OUTLAYS: SATELLITE + RECTENNA + LAUNCH + CONSTRUCTION**
- **THE ANNUAL AMOUNT PAID IS (FOR ELECTRIC UTILITIES) EXPRESSED AS INITIAL COSTS TIMES A "CAPITAL CHARGE FACTOR"**
- **$\$/\text{kWh} = \frac{(\text{INITIAL COSTS}) \times (\text{CAPITAL CHARGE FACTOR})}{\text{kWh PRODUCED IN YEAR}}$**
- **THUS AVAILABILITY IS AS IMPORTANT AS INITIAL COSTS IN DETERMINING COMMODITY PRICE**
- $\text{AVAILABILITY} = \frac{\text{kWh PRODUCED IN YEAR}}{(\text{PLANT CAPACITY}) \times (8,766 \text{ HOURS})}$**
- **HENCE A DEMONSTRATION OF THE ACHIEVEMENT OF REQUIRED AVAILABILITY IS AS IMPORTANT AS THE DEMONSTRATION OF ACHIEVEMENT OF REQUIRED COSTS**

**D180-24071-3**

### **SPS COST FACTORS**

The projections of cost from a precursor satellite to a full sized unit are relatively straightforward. Learning factors can be used to relate initial unit costs to the cost of hardware and mass production.



SPS-1836

D180-24071-3

## SPS Cost Factors

**BOEING**

### ACQUISITION COSTS

|                        |                                                         |
|------------------------|---------------------------------------------------------|
| • RECTENNA             | SITE, SITE PREP., MATERIALS FABRICATION<br>INSTALLATION |
| • SATELLITE HARDWARE   | MATERIAL, FABRICATION                                   |
| • ORBITAL PLACEMENT    | SITES, FACILITIES, FLEET, PROPELLANTS,<br>OPERATIONS    |
| • ORBITAL CONSTRUCTION | CONSTRUCTION BASE, OPERATIONS                           |

### OPERATIONS COSTS

|                                 |                               |
|---------------------------------|-------------------------------|
| • OPERATIONS AND<br>MAINTENANCE | LOGISTICS, SPARES, OPERATIONS |
|---------------------------------|-------------------------------|

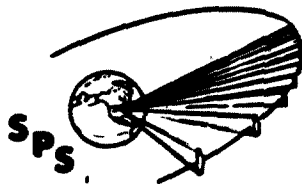
### DEVELOPMENT

- SIGNIFICANT IN INVERSE PROPORTION TO THE PROGRAM SIZE

### DEMONSTRATION OF COMMERCIAL VIABILITY

In order to show that full size solar power satellites can achieve a high plant factor, that is high percentage of potential use, it will be necessary to demonstrate good availability for the precursor unit. To do this the precursor unit can be operated for a period of time and the availability can be recorded. From this demonstrated availability statistical analyses can be used to compute the probable availability of a full size unit. Whenever availability or reliability are calculated it is necessary to express an associated level of confidence. If a statement is made that the expected reliability is 0.95, for example, and the confidence in this factor is 0.9, what we are saying is that there is only one chance in ten that the eventual reliability will be less than 0.95. Even a relatively small precursor satellite will have sufficient components to allow a test of perhaps of one year duration to have sufficient statistical significance. Even with extremely good reliability for such components as klystron transmitter tubes a SPS program would not have commercial viability unless effective maintenance method can be developed. Maintenance of the precursor satellite should therefore be demonstrated.

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SPS-1818

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# Demonstration of Commercial Viability

SOEING

## AVAILABILITY

- **DEMONSTRATE NECESSARY AVAILABILITY (RELIABILITY)**
  - 1) **OPERATE A PRECURSOR SPS UNIT FOR A PERIOD AND MEASURE THE AVAILABILITY**
  - 2) **USE STATISTICAL ANALYSES TO COMPUTE THE PROBABLE AVAILABILITY OF A FULL SIZE UNIT**
- **THE LEVEL OF CONFIDENCE IN THIS AVAILABILITY IS INFLUENCED BY:**
  - **DURATION**

**HAS STATISTICAL SIGNIFICANCE: ALSO PROVIDES OPPORTUNITY FOR UNEXPECTED EVENTS**
  - **SIZE OF UNIT**

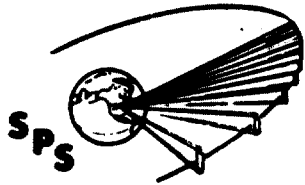
**THE NUMBER OF COMPONENTS HAS STATISTICAL SIGNIFICANCE: A UNIT WITH 100 KLYSTRONS WOULD REQUIRE A LONGER TEST THAN ONE WITH 1,000 KLYSTRONS (FOR THE SAME CONFIDENCE)**
- **DEMONSTRATION OF MAINTAINABILITY IS OF OBVIOUS NECESSITY IN SHOWING AVAILABILITY**

**MAINTAINABILITY INVOLVES: FAULT DETECTION, SPARES, LOGISTICS, REPLACEMENT, CHECKOUT, ETC.**

**PERFORMANCE ASSURANCE: SATELLITE POWER GENERATION**

Critical parameters for the power generation portion of the SPS are listed here, along with the demonstrations which is probably required for those parameters. For example, solar array efficiency can probably be completely verified on small solar blanket panels. The exact geosynchronous radiation environment, including electrons and protons with broad energy spectra, is probably quite difficult to quantify and even more difficult to exactly simulate. Hence, long term exposure of solar array panels to this environment is probably required in order to have full assurance of eventual SPS operational factors. If a self power transfer from a low assembly orbit to geosynchronous orbit is baselined, and it currently appears that this is the most economic approach, then the precursor demonstration should include exposure of solar array panels in a self power transit. If annealing is to be part of the operational satellite program then it should also be demonstrated in the precursor program using solar array panels which had been degraded as a result of exposure to the actual self power transfer and geosynchronous environment.





SPS-1838

D180-24071-3

# Performance Assurance Satellite Power Generation

BOEING

## PERFORMANCE

1. SOLAR ARRAY EFFICIENCY  
\_\_\_\_\_ %
2. SOLAR ARRAY RADIATION  
RESISTANCE  
\_\_\_\_\_  
\_\_\_\_\_  
\_\_\_\_\_
3. SOLAR ARRAY ANNEALABILITY
4. POWER DISTRIBUTION: CELL  
STRINGS  
\_\_\_\_\_ I2R  
\_\_\_\_\_ DAMAGE  
\_\_\_\_\_ SUSCEPTABILITY

## DEMONSTRATION

MEASUREMENTS ON SMALL PANELS (M)<sup>2</sup>.

EXPOSURE TO THE GEOSYNCHRONOUS  
(AND ORBIT TRANSFER, IF LEO  
ASSEMBLY) ENVIRONMENT FOR AN  
APPROPRIATE PERIOD (TO INCLUDE  
SUB-STORMS AND FLARES).  
POSSIBLE ADDITIONAL CHAMBER  
TESTS.

THE FULL  
ENVIRONMENT  
IS DIFFICULT TO  
MEASURE/  
SIMULATE.

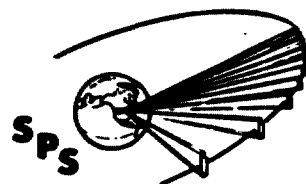
MEASURE ARRAY PERFORMANCE BEFORE AND AFTER  
ANNEALING (BY THE "OPERATIONAL" METHOD).  
DAMAGE TO HAVE BEEN CAUSED BY ACTUAL GEO-  
SYNCHRONOUS ENVIRONMENT (PLUS TRANSFER FROM  
LEO IF SELF POWER IS TO BE USED)

FULL HIGH VOLTAGE  
SERIES-PARALLEL  
STRINGS-PLASMA INTERACTIONS, CHARGING

**PERFORMANCE ASSURANCE: SATELLITE POWER GENERATION (CONTINUED)**

This chart continues the list of critical power generation parameters along with the probable required demonstration for each. Satellite dynamics are probably most critical as regards pointing of the microwave power transmitter. A preliminary pointing tolerance for this transmitter is one minute of arc, mechanical. The eventual full size transmitting antenna will probably incorporate its own attitude control system including control moment gyros and yoke drives. These critical elements should be demonstrated on the precursor unit, since this relates not only to commercial viability of the solar power system but also to environmental aspects related to the pointing of the microwave beam.

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SPS-1830

D180-24071-3

## Performance Assurance Satellite Power Generation (Contd)

BOEING

### PERFORMANCE

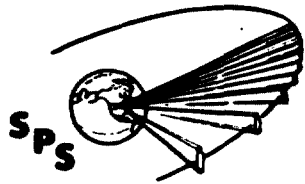
5. POWER DISTRIBUTION:  
BUSSES  
 $I^2R$  LOSS < TBD%
6. ROTARY JOINT  
SLIP RING WEAR
7. SYSTEM DYNAMICS  
(INCLUDING POINTING,  
SMOOTHNESS OF MOTION,  
INTERACTION WITH REST  
OF SPS)
8. STRUCTURAL PROPERTIES

### DEMONSTRATION

- DESIGN ANALYSIS  
ELEMENT TEST  
PLASMA INTERACTIONS  
CHARGING
- ELEMENT TEST
- OPERATION OF SUB-SCALE SPS SYSTEM. SYSTEM TO INCLUDE POSITION SENSORS ON BOTH TRANSMITTER AND POWER GENERATION SYSTEM, YOKE DRIVES, THRUSTERS ON POWER GEN SYSTEM, CMGS (AND/OR OTHER CONTROLLERS) ON ANTENNA, CONTROL COMPUTER, ETC. . MAINTAIN REQUIRED POINT ( $\pm 1$  MINUTE, PRELIM) OVER A PERIOD OF TIME WHICH ALLOWS SIGNIFICANT ACCUMULATION OF METEOROID INDUCED MOMENTUM.
- ACCELERATED ENVIRONMENTAL EXPOSURE  
PROPERTIES TEST

### PERFORMANCE ASSURANCE: TRANSMITTER

The transmitter waveguides require high dimensional accuracy and stability. The waveguides for the precursor unit should be produced by the methods expected for the full size satellite and should be transported and assembled by similar methods, to demonstrate that the eventual mechanical flatness of the antenna can be achieved. Continuous high accuracy operation of the microwave power transmission phase loop control system should also be accomplished. Critical elements here are the successful conjugation of the reference frequency and the transmitter frequency by the antenna electronics, distribution of the reference frequency from the central generator to all parts of the antenna with compensation for changes in length of this distribution path, the frequency offset of the pilot beam from the power beam, the reception of the pilot beam by the distributed receivers of the spacetenna, the reliability of these receivers and the ground transmitter and phase control system. A long term demonstration of successful operation should include various weather conditions including rain, Faraday rotation of the ionosphere and other ionospheric variations. A critical reliability element for the transmitter is of course the klystron tubes. Current klystron reliability would be inadequate for the SPS. During the SPS development program this reliability must be increased. The demonstrator programs would operate a sufficient quantity of tubes for a sample period long enough to get through their "infant mortality" period and provide, with the required confidence, a test of the eventual reliability. Of course full reliability may not have been achieved by the time of precursor program. However, the achieved reliability could be correlated with historical trends of increase in reliability to indicate eventual operation with the required reliability in the full size program.



SPS-1836

D180-24071-3

# Performance Assurance Transmitter

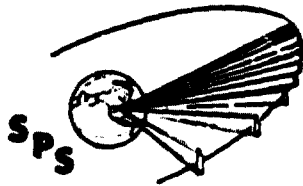
BOEING

- |                                                      |                                                                                                                                                                                                                                                                                              |
|------------------------------------------------------|----------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| <b>1. WAVEGUIDE</b>                                  | <b>DIMENSIONAL ACCURACY AND STABILITY OF SIGNIFICANT SAMPLES OF WAVEGUIDE PANELS, MANUFACTURED AND ASSEMBLED IN THE "OPERATIONAL" METHOD.</b>                                                                                                                                                |
| <b>2. STRUCTURAL STABILITY</b>                       | <b>SUBARRAY DEPLOYMENT PRECISION AND STABILITY WITH FULL THERMAL LOAD</b>                                                                                                                                                                                                                    |
| <b>3. PHASE LOOP OPERATION</b>                       | <b>CONJUGATION, DISTRIBUTION<br/>PILOT BEAM OFFSET—RECEIVE PILOT VIA SPACETENNA<br/>RECEIVER RELIABILITY, GROUND TRANSMITTER<br/>LONG TERM DEMONSTRATION, VARIOUS WEATHER CONDITIONS,<br/>IONOSPHERIC CONDITIONS (BEAM "WIDTH EFFECTS"?)</b>                                                 |
| <b>4. THERMAL INTEGRITY</b>                          | <b>DEMONSTRATION OF MULTIPLE PANEL SYSTEM AT VARIOUS POWER LEVELS</b>                                                                                                                                                                                                                        |
| <b>5. KLYSTRON PERFORMANCE INCLUDING RELIABILITY</b> | <b>1. OPERATE A SAMPLE FOR A PERIOD OF TIME SUFFICIENTLY LONG TO:<br/>a. GET THROUGH INFANT MORTALITY<br/>b. PROVIDE DESIRED CONFIDENCE<br/><br/>2. CORRELATE WITH PRODUCTION RATE AND INDUSTRIAL MATURITY AND HISTORICAL IMPROVEMENT TRENDS<br/><br/>3. MEASURE EFFICIENCY—SMALL SAMPLE</b> |

### ESTABLISHING A MTBF INTERVAL

This chart shows how various levels of meantime between failure intervals can be demonstrated based on a variable period of operation and the observed failure rate. For example, if a meantime between failure of 20 years or more is required the upper of the two criterion lines applies. If a plot of test results, that is fractional failures vs time, falls above this line it can be said that the meantime between failures is either 18.48 years with a confidence level of 0.99 or, for example, 20 years with a confidence level of 0.8. The lower of the two lines indicates lower MTBF intervals. For example if the actual test results fall along this line a MTBF of only 5.07 years with a confidence level of 0.8 would be indicated. This is probably the lowest MTBF for commercial acceptability of the SPS, since this could lead to plant factors as low as 0.6.

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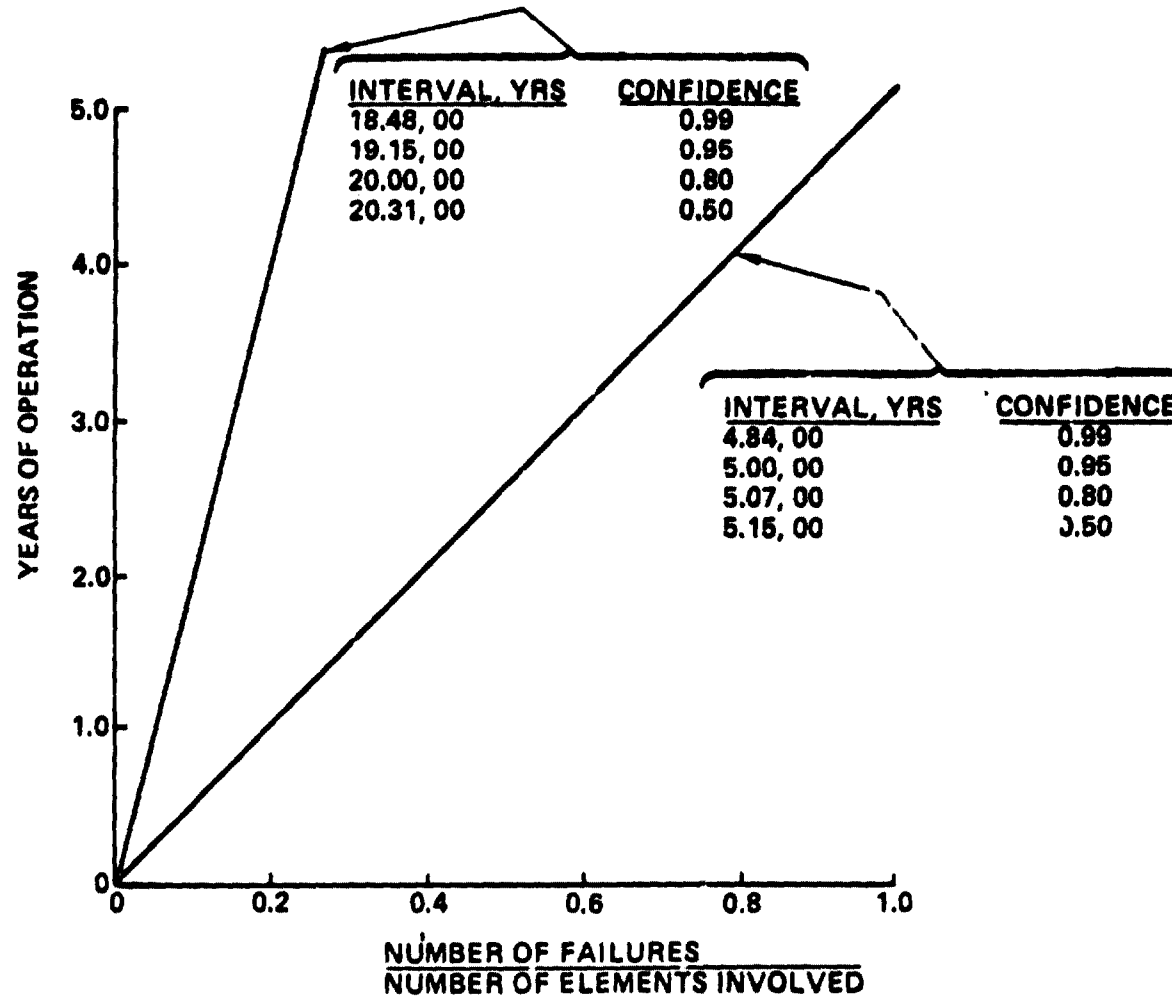


SPS-1821

D180-24071-3

# Establishing a MTBF Interval

BOEING

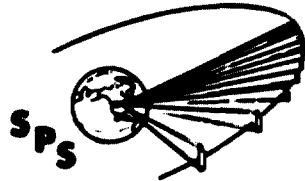


(FOR SAMPLES  $\leq 500$ )

**PERFORMANCE ASSURANCE: RECTENNA**

Critical rectenna performance parameters are listed along with the demonstration probably required to assure that the required levels will be present in the full sized system. For example, the conversion of the RF energy to direct current is probably demonstratable on a few production panels (perhaps only several hundred square meters). The 89% figure here given is probably the conversion efficiency achievable at the center of the microwave beam; the 80% number is close to that achievable at approximately 1 milliwatt per square centimeter at the edge of the microwave beam. Attenuation here relates to the value of microwave energy which is present beneath the rectenna panels. This probably is critical if multiple land use is envisioned, for example agriculture performed beneath the rectenna. Full long-term environmental resistance is probably the most difficult of the rectenna characteristics to demonstrate. Rectenna panels could be tested in a large weather chamber, for example, the type available at Eglin Air Force Base. This, coupled with a field test, would probably be sufficient to demonstrate adequate environmental resistance to proceed with at least 1 or 2 full-sized rectennas.





SPS-1840

D180-24071-3

# Performance Assurance Rectenna

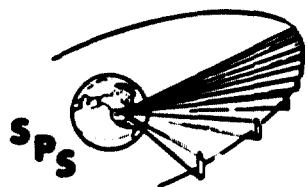
BOEING

| <u>PERFORMANCE</u>                                                                                          | <u>DEMONSTRATION</u>                                         |
|-------------------------------------------------------------------------------------------------------------|--------------------------------------------------------------|
| 1. <u>CONVERSION EFFICIENCY</u><br>89% AT 23 mW/cm <sup>2</sup><br>80% AT 1 mW/cm <sup>2</sup>              | MEASUREMENTS ON PANELS OF<br>APPROXIMATELY 100M <sup>2</sup> |
| 2. <u>ATTENUATION</u><br>(BEAM STRENGTH BELOW<br>ANTENNA) TBD dB                                            | MEASUREMENTS ON PANELS OF<br>APPROXIMATELY 100M <sup>2</sup> |
| 3. <u>ENVIRONMENTAL RESISTANCE</u><br>RAIN _____<br>SNOW _____<br>WIND _____<br>HAIL _____<br>SEISMIC _____ | WEATHER CHAMBER TEST<br><br>(FIELD TEST?)                    |
| 4. <u>POWER DISTRIBUTION</u><br>I <sup>2</sup> R LOSS < TBD%                                                | DESIGN ANALYSIS,<br>ELEMENT TEST                             |
| 5. <u>POWER CONVERSION</u><br>DC → AC OR DC → DC<br>EFFICIENCY > %                                          | COMPONENT TEST<br>(CONVERTER OR MOTOR GENERATOR)             |
| 6. <u>SWITCHING, CROWBAR</u><br>OPEN CIRCUIT, TRANSIENT<br>PROTECTION                                       | COMPONENT TEST AND TEST<br>WITH RECTENNA PANEL               |

### **PHOTOVOLTAIC REFERENCE CONFIGURATION**

This chart shows a plan and side view of a full size SPS having a 5,000 megawatt output at each of the two rectennas. The power generation portion of the satellite is divided into 8 bays which are transferred individually from the low assembly orbit to geosynchronous orbit using the self-power method. These 8 modules are each composed of 32 identical bays. It is possible to envision a precursor SPS as being made up of 1, 2 or more bays up to a full 32, that is 1 module. Thus the precursor design would be directly relatable to the full size system. The transmitter would be a subscale unit with a power transmission capability appropriate to the number of bays provided. The transmitter can be made up of full size antenna components. The precursor SPS receiving major attention in this study was one composed of 4 bays. This will be given on subsequent charts. Such a precursor allows the use of full, exact scale, solar array panels which generate the full required voltage over the same path link with the same physical arrangement.

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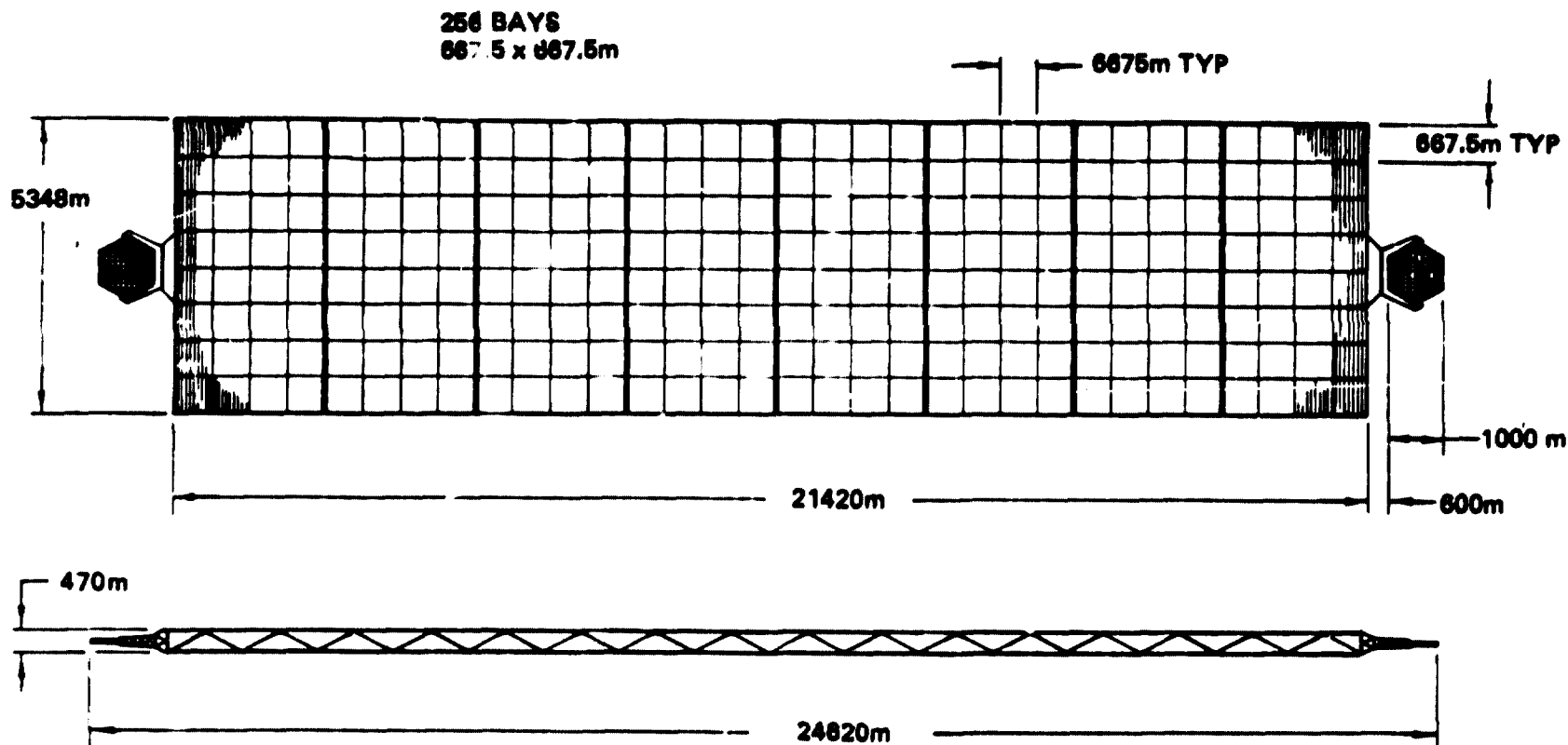


SPS-1881

D180-24071-3

# Photovoltaic Reference Configuration (5,000 MW Output Each Transmitter)

BOEING

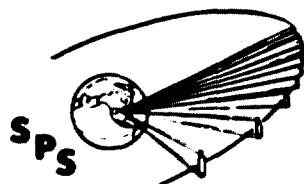


|                        |                               |
|------------------------|-------------------------------|
| TOTAL SOLAR CELL AREA: | 101.8 km <sup>2</sup>         |
| TOTAL ARRAY AREA:      | 110.2 km <sup>2</sup>         |
| TOTAL SATELLITE AREA:  | 114.5 km <sup>2</sup>         |
| OUTPUT:                | 16.93 GW MINIMUM TO SLIPRINGS |

### SUITABILITY OF A SMALL PRECURSOR

A precursor unit composed of 4 bays will have 1.56% of the area of the full size SPS. Hence, the power generated by such a precursor would be approximately 1.5% of that of the full size satellite. This chart addresses the question of whether or not such a satellite is adequate to demonstrate commercial viability. Some of the critical developmental parameters of the satellite are listed along with their relationship to the full size satellite. For example, the solar cells can be of the same basic type and same thickness, i.e., 50 micrometers, (2 mils) although the cells may not have achieved the full efficiency of those to be used in the eventual satellite. The solar cell blankets can have the same physical parameters as those for the eventual full size SPS as regards cover type, thickness, substrate, cell interconnects, etc. The structural elements of the precursor can be exactly those to be used on the full sized system, if the launch vehicle has the capacity to carry up either the required beam machines, the structural components, or both. Solar cell costs are of course critical to the eventual commercial viability of this system. The number of solar cells required for a 4 bay precursor is sufficiently large to warrant development of the full production capacity to be used in the SPS system. That is, perhaps 1 or more production lines, to achieve a capacity of approximately 20,000 cells per hour, the rate required to make the cells for the precursor in a period of two years. For the full size satellite, more of these production lines would be used to obtain the production rate of 2 million cells per hour which goes with a satellite addition rate of 1 per year. For the transmitter the subarrays can be of the type and size to be used in the full sized system. The klystron tubes can be full power tubes, that is, 72 kilowatts, however, in the precursor the tube mass and efficiency may not meet the target goals required for the full size system. Other transmitter elements can closely parallel those to be used in the full size SPS. Rectennas can be smaller area versions of the full size system and use panels, mounting methods, switch gear, etc. of the final design. A self-powered transport to geosynchronous orbit can be used with power processors and thrusters of the type to be employed on the full size SPS. Their efficiency might be slightly lower than that for the eventual system.

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SPS-1837

D180-24071-3

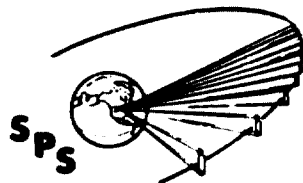
# Suitability of a "Small" Precursor

BOEING

| <u>DEVELOPMENT</u>        | <u>RELATIONSHIP TO FULL SIZE</u>                | <u>1.58% SPS<br/>ADEQUATE</u> |
|---------------------------|-------------------------------------------------|-------------------------------|
| <b>POWER GENERATION</b>   |                                                 |                               |
| CELL                      | SAME THICKNESS, SOMEWHAT LOWER EFFICIENCY       | ✓                             |
| BLANKET                   | SAME                                            | ✓                             |
| VOLTAGE, CURRENT          | SAME                                            | ✓                             |
| STRUCTURE                 | SAME (15M WIDTH, JOINTS, MATERIAL)              | ✓                             |
| ATTITUDE CONTROL          | SAME (PERHAPS LOWER EFFICIENCY)                 | ✓                             |
| PRODUCTION EQUIPMENT      | SAME (16,000 CELLS PER HOUR, TWO YEARS)         | ✓                             |
| ROTARY JOINT              | SMALLER DIAMETER, SAME CONTACT SYSTEM           | ✓                             |
| SWITCHGEAR                | SAME (PERHAPS SUBSCALE)                         |                               |
| RELIABILITY DEMONSTRATION | YES, ENOUGH COMPONENTS                          |                               |
| <b>POWER TRANSMISSION</b> |                                                 |                               |
| TUBE                      | SAME (70KW)(PERHAPS HEAVIER, LOWER MTBF)        | ✓                             |
| WAVEGUIDE                 | SAME                                            | ✓                             |
| STRUCTURE                 | SAME (FULL DEPTH)                               | ✓                             |
| PHASE CONTROL             | SAME (EXCEPT 1/5 DISTANCE FOR REF. PHASE DIST)  | ✓                             |
| PRODUCTION EQUIPMENT      | SAME (NEED ABOUT 5000 TUBES)                    | ✓                             |
| SWITCHGEAR                | SAME                                            | ✓                             |
| ATTITUDE CONTROL          | SAME (PERHAPS SUBSCALE CME's)                   | ✓                             |
| RELIABILITY DEMONSTRATION | YES, ENOUGH COMPONENTS                          | ✓                             |
| <b>POWER RECEPTION</b>    |                                                 |                               |
| RECTENNA                  | SAME (AT LOWER EFFICIENCY, PERHAPS 60% NOT 85%) | ✓                             |
| <b>LEO TRANSPORT</b>      | GOOD (WITH FLYBACK BOOSTER)                     | ✓                             |
| <b>GSO TRANSPORT</b>      |                                                 |                               |
| THRUSTER                  | SAME OR SUBSCALE                                | ✓                             |
| POWER PROCESSOR           | SAME OR SUBSCALE                                | ✓                             |

**SUITABILITY OF A SMALL PRECURSER CONSTRUCTION BASE**

Analyses of potential construction methods for a 1 bay to 32 bay size precursor satellite have indicated that a one-eighth scale construction base would be appropriate. This one-eighth scale construction base would closely approximate a segment of a full size construction base. It would have the capability to produce the full depth bays to be used in the full sized satellite. The chart shows that each of the critical construction base parameters are demonstratable on this one-eighth scale unit. As in the full size construction process, the antenna is built separately and mated to the yoke and the power generation module. The four power generation bays would be built in two groups of two, forming two modules which individually self-power to the operational orbit. One of these modules would carry the transmitter. In the operational orbit the two modules would be berthed together in a process similar to that to be used in the full sized system.



SPS-1813

D180-24071-3

# Suitability of "Small" Precursor (Construction Base)

**BOEING**

| <u>DEVELOPMENT</u>     | <u>RELATIONSHIP TO FULL SIZE</u>                      | <u>WITH "12% BASE"</u> |
|------------------------|-------------------------------------------------------|------------------------|
| <b>CONSTRUCTION</b>    |                                                       |                        |
| BEAM MACHINE           | SAME (EXCEPT TRANSPORT PACKAGING)                     | ✓                      |
| BEAM INTERCONNECTS     | SAME                                                  | ✓                      |
| SOLAR ARRAY DEPLOYMENT | 2X7.5m=15M ("CANISTER BOOSTER") 21X3.75=15M (ORBITER) | ✓                      |
| BUSBAR DEPLOYMENT      | SAME, BUT SUBSCALE                                    | ✓                      |
| MODULE "INDEXING"      | SAME                                                  | ✓                      |
| ASSEMBLE ANT. STRUC.   | SAME                                                  | ✓                      |
| DEPLOY ANT. SUBARRAYS  | SAME                                                  | ✓                      |
| MATE ANT. TO YOTE      | SAME                                                  | ✓                      |
| MODULE BERTHING        | SAME                                                  | ✓                      |
| ANTENNA XPORT LOCATION | SAME                                                  | ✓                      |

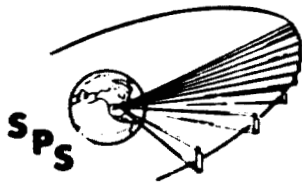
**PRECURSOR SPS**

This chart shows the four bay precursor SPS including its transmitter system. Each of the four bays is identical to that to be used in the full SPS. The 470 meter depth of this unit is exactly that, for example, of the full size system. The left-hand two bays form one module which is constructed first and dispatched to geosynchronous orbit, then the right-hand two modules and the transmitter are constructed and dispatched. Each of the two modules mounts self-power transfer electric thrusters at its four corners on extensions of the 15 meter structural beams. The antenna is displaced from the edge of its power module by a structural system which allows the antenna to be rotated to a position beneath the center of that module. This method is used in the full size system, where the transmitter antenna is rotated to a position beneath the center of the module. The 190 meter diameter transmitter is composed of full sized subarrays arranged so as to provide the requisite power taper. In the center of the transmitter, nine subarrays of the full power type are grouped so as to duplicate the maximum thermal environment of the full size transmitter. Flat sheet aluminum busbar conductors route the power from the solar arrays to the rotary joint and yoke system.

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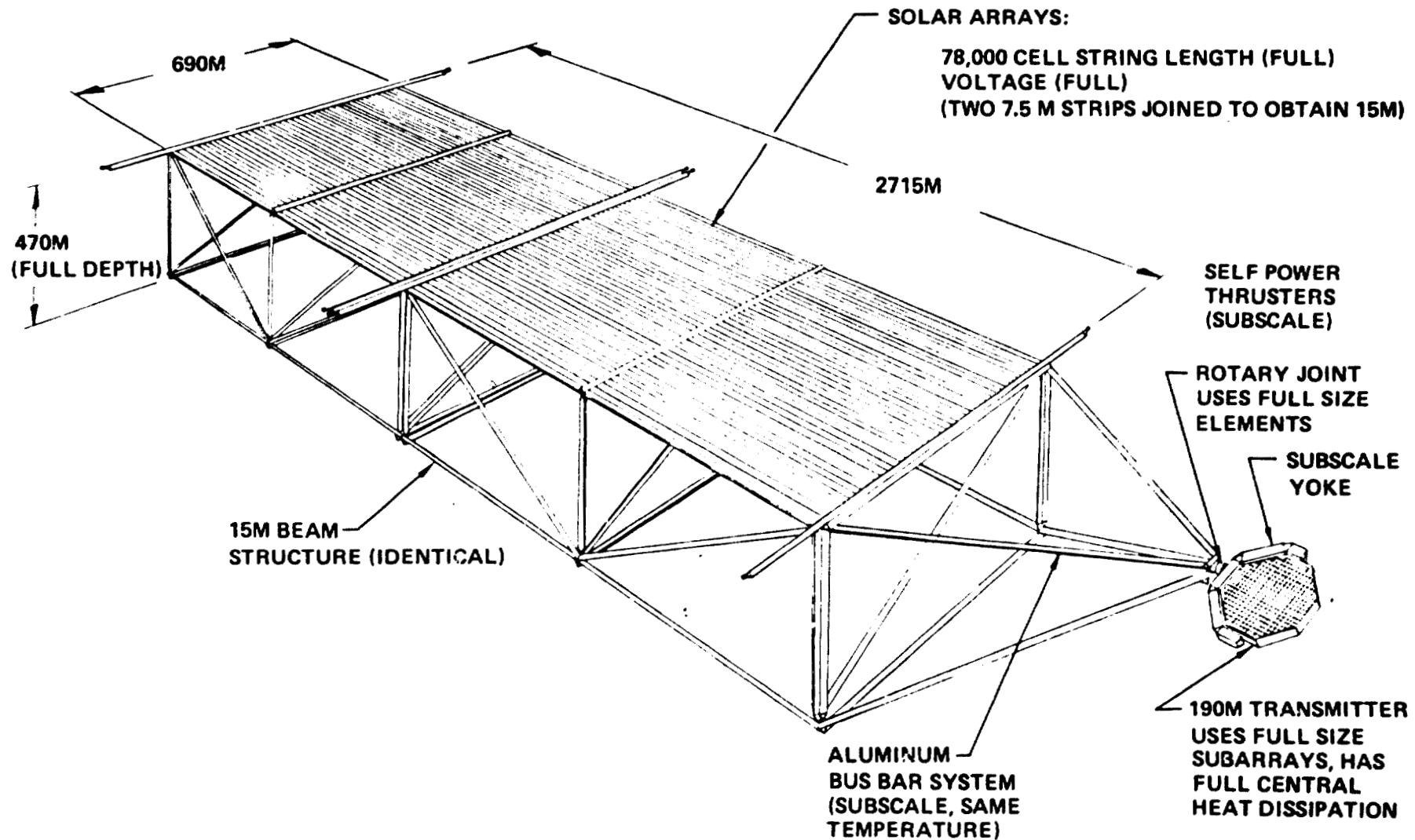


# PRECURSOR SPS



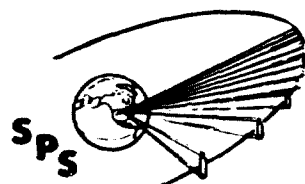
SPS-1812

BOEING



**POWER TRANSMITTED**

If a precursor satellite is to be operated in the later part of the 1980's the solar cells for that satellite must undergo a design freeze in approximately 1983. The chart shows how the natural progression from today's solar cells to the high efficiency solar cells of the full size SPS will result in an efficiency of approximately 16% for the precursor cells. With this efficiency and with the probable transmitter parameters achievable at the time of the precursor satellite, approximately 185 megawatts will be launched from the face of the transmitter. This is approximately 3% of the power of the full size transmitter.



0180-74071-3

## Power Transmitted

SPS-1826

BOEING

### • SOLAR ARRAY PERFORMANCE: (50 $\mu$ M CELLS)

| 1977        |                       | 1983                    | 1990        |
|-------------|-----------------------|-------------------------|-------------|
| OCLI        | 11.2 TO 12.3%         | ESTIMATES OF            | 15.75% CELL |
| SPECTROLAB  | 10.3 TO 11.3%         | HELIOTEK & H. OMAN      | 17.33% WITH |
| SOLAREX     | 11.2%                 | 14.5% BARE CELL         | TEXTURING & |
| ASSUMPTION: | 75 $\mu$ M VEE-GROOVE | YIELDS 16% WITH         | VEE-GROOVES |
|             | COVERS WILL ALLOW     | TEXTURING & VEE-GROOVES |             |
|             | A 12% CELL TO         |                         |             |
|             | OBTAIN 13%            |                         |             |

### • TRANSMITTER OUTPUT

|                                                         |        |
|---------------------------------------------------------|--------|
| OUTPUT OF FOUR FULL SIZE SPS "BAYS," WORST ILLUMINATION | 289 MW |
| WITH 1983 CELLS: (16%/17.33% x 289)                     | 267 MW |
| BUSBAR I <sup>2</sup> R, 0.98 (SHORTER THAN FULL SIZE)  | 261 MW |

|                                |              |                          |
|--------------------------------|--------------|--------------------------|
| ANTENNA POWER DISTRIBUTION     | 0.98         | (SHORTER THAN FULL SIZE) |
| DC $\rightarrow$ RF CONVERSION | 0.82         | (0.85, ULTIMATE)         |
| WAVEGUIDE I <sup>2</sup> R     | 0.985        | FULL SPS                 |
| IDEAL BEAM                     | 0.965        | FULL SPS                 |
| INTER SUBARRAY                 | 0.946        | FULL SPS                 |
| <u>INTRA SUBARRAY</u>          | <u>0.981</u> | <u>FULL SPS</u>          |
|                                | 0.71         |                          |

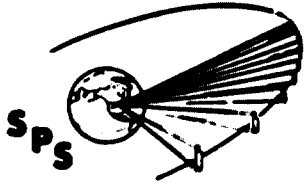
HENCE 185 MW ARE "LAUNCHED"

FULL SIZE SPS IS 6220 MW LAUNCHED (PER TRANSMITTER)  $185/6220 = 0.03$

**BEAM PATTERN**

This chart relates attenuation of the microwave beam to the radial distance from the center of that beam. The central beam strength of the full size power satellite is approximately 23 milliwatts or 23,000 microwatts. The chart shows a simple relationship for determining the central beam strength developable by the precursor unit. The 4-bay precursor with a 190 meter diameter transmitter achieves a central beam strength of slightly over 23 microwatts per square centimeter or 1/1,000 of the beam strength of the full size system. At a radius of approximately 10 miles, the beam strength diminishes to approximately 10 microwatts per square centimeter which is the Soviet continuous exposure standard. Thus, the beam diameter of 20 miles is relatable to the tightest exposure standard in the world.

# Beam Pattern



SPS-1827

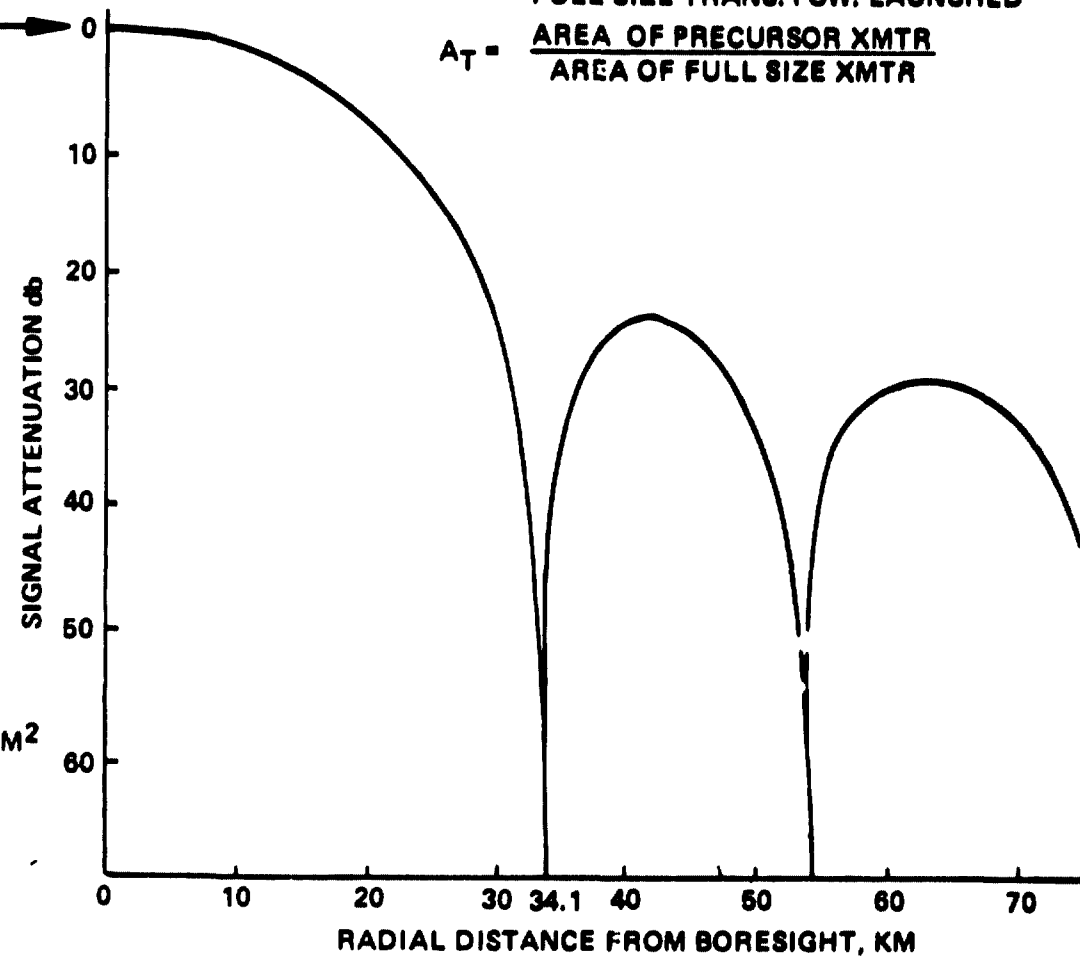
BOEING

$P_{MAX}$  = CENTRAL BEAM STRENGTH

$P_T$  =  $\frac{\text{POWER LAUNCHED BY TRANS}}{\text{FULL SIZE TRANS. POW. LAUNCHED}}$

$A_T$  =  $\frac{\text{AREA OF PRECURSOR XMTR}}{\text{AREA OF FULL SIZE XMTR}}$

$$P_{MAX} = 23,000 \mu\text{W}/\text{CM}^2 \times (P_T) \times (A_T)$$



## REFERENCE PRECURSOR

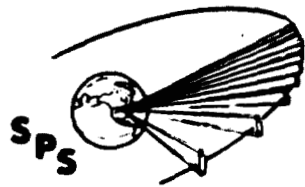
•  $P_T = 0.03$  (OF FULL SIZE)

•  $A_T = 0.036$  (OF FULL SIZE)

$$P_{MAX} = (23,000)(0.03)(0.036) = 24.6 \mu\text{W}/\text{CM}^2$$

### **RECTENNA OPTIONS**

It is probably not appropriate to construct a full area rectenna for the precursor SPS. With the full rectenna area a power output of approximately 85 megawatts would be developed by the 4-bay precursor SPS studied. This power level is of little commercial significance relative to the cost of the precursor program. A partial rectenna, as shown, with an area of 7 square kilometers would probably produce about 1 megawatt with the precursor's central beam strength. This 1 megawatt is probably quite adequate to demonstrate successful integration of space generated power into a commercial utility network. The beam area around the rectenna section would be instrumented in order to determine the characteristics of the transmitted beam.

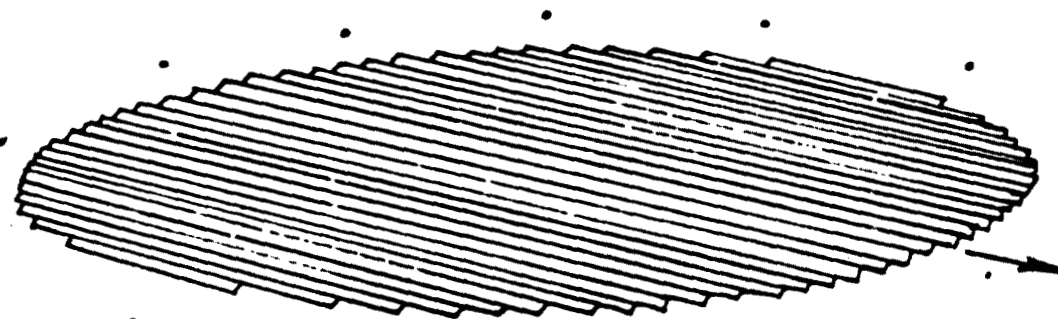


SPS-1819

D180-24071-3

## Rectenna Options (For "1.56 % PSPS")

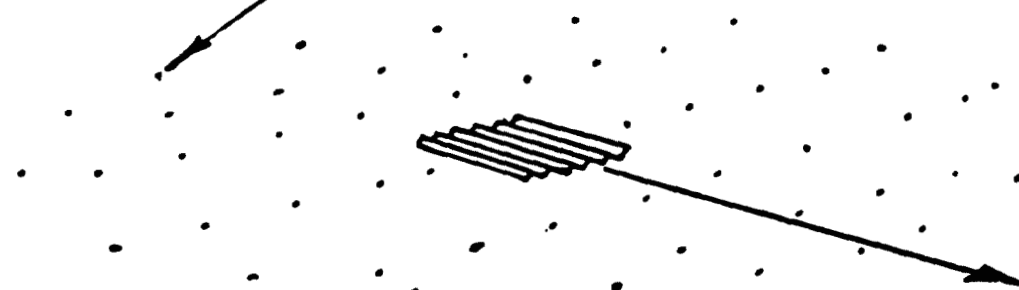
BOEING



### "FULL BEAM AREA"

- DIA. = 20 km (12 mi)
- OUTPUT  $\approx$  85 MW

INSTRUMENTATION SITES



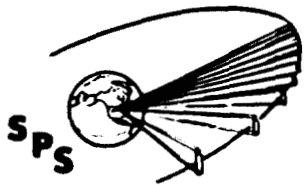
### "PARTIAL"

- 2.6 km x 2.6 km (1.7x1.7mi)
- OUTPUT  $\approx$  1 MW

**RECTENNA CONSIDERATIONS**

It is probably appropriate to direct the microwave beam from the precursor satellite to a government reservation. If the most stringent current microwave exposure standards is used, and such a standard may be levied upon the solar power project before it has demonstrated long term successful microwave beam pointing, then we will be working with a standard of 10 microwatts per square centimeter; at and above this power level the precursor beam diameter is 20 miles. This will fit conveniently within a government reservation such as White Sands Proving Ground, New Mexico and would be conveniently near the power distribution network at Alamogordo, New Mexico. Despite the low central beam strength and the lower efficiency of rectennas at these levels a precursor rectenna of approximately 7 square kilometers, that is, 1.7 miles by 1.7 miles, would be adequate to develop one megawatt and would cost approximately 5% of the cost associated with the total precursor program.





SPS-1833

D180-24071-3

## Rectenna Considerations

**BOEING**

- GOVERNMENT RESERVATION DESIRED
  - CONTAIN APPROXIMATELY 20 MILE DIAMETER BEAM SECTION WHICH IS ABOVE SOVIET STANDARD ( $10 \mu\text{W}/\text{cm}^2$ )
  - SIGNAL LEVEL ( $\sim 23 \mu\text{W}/\text{cm}^2$ ) MAXIMUM WOULD PERMIT PERSONNEL EXPOSURE
- LOCATE NEAR EXISTING TRANSMISSION LINE
  - INSERT ONE MEGAWATT
- RECTENNA COULD USE "FINAL SPS" COMPONENTS
- EVEN AT ONLY 60% EFFICIENCY,  $7 \text{ km}^2$  (1.7 MILE x 1.7 MILE) PRODUCES 1 MW
- AT  $\$100/\text{m}^2$ , RECTENNA FOR 1 MW IS "ONLY"  $\$0.7\text{B}$

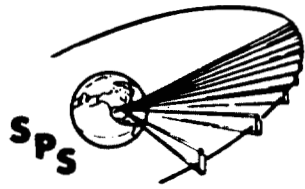
**PRECURSOR SPS CONSTRUCTION BASE**

This chart shows the configuration of the precursor SPS LEO construction base. Comparison of this facility to that shown for the full-size SPS LEO base reveals that the precursor facility is a one-bay corner of the full size facility.

The modules and the yoke would be constructed in the larger part of the base using the full-size construction equipment that would be used to construct the full size SPS modules. The significant operational difference would be that after the frame for one of the module bays is constructed the frame assembly equipment would be moved out of the way so that the solar array deployment machine could be moved into the same construction bay. After the array is deployed, the completely assembled bay is indexed out onto the outriggers so that the second bay could be assembled.

The yoke and thruster systems would also be constructed in this bay.

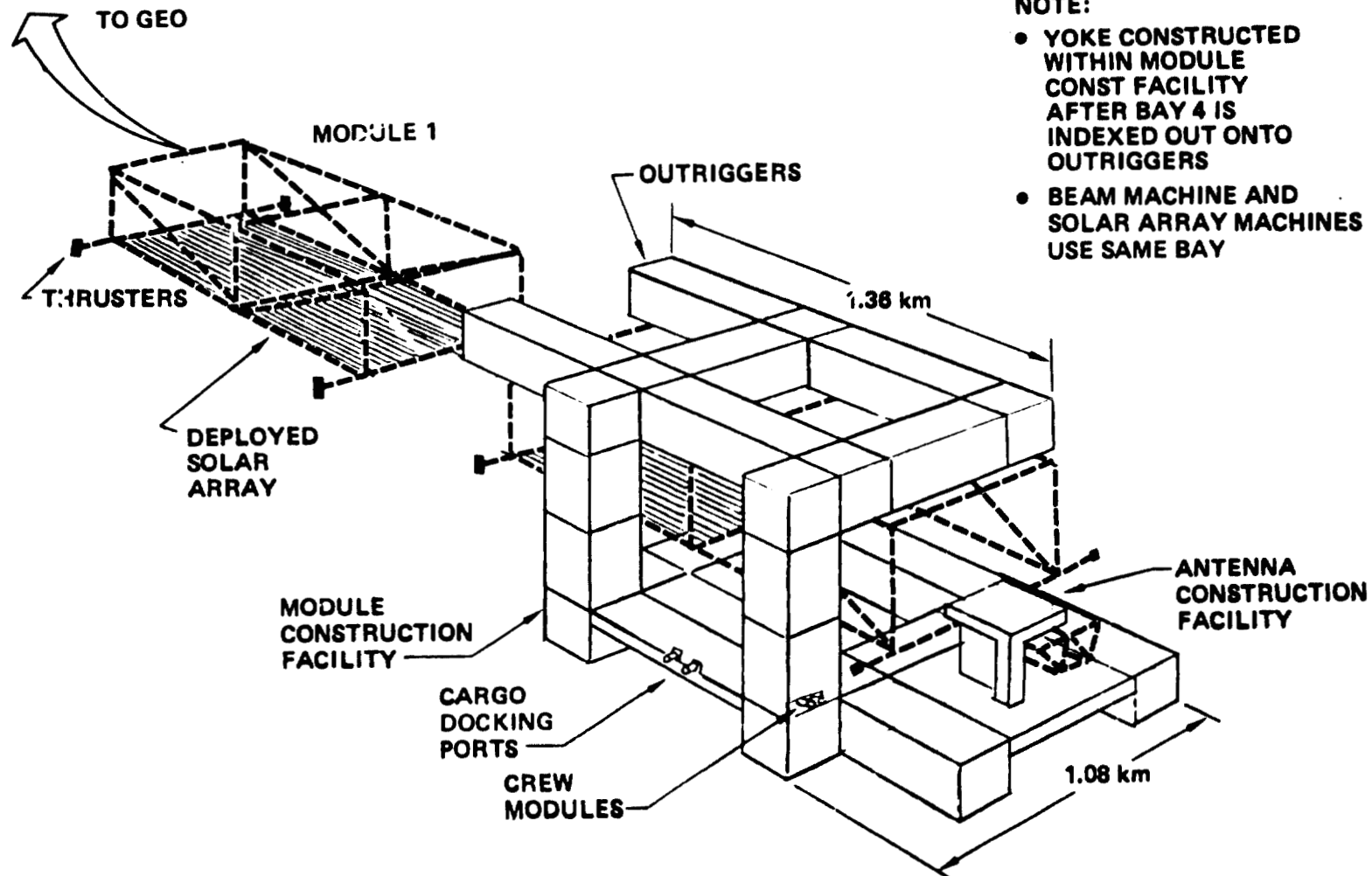
The antenna would be constructed in the antenna construction facility which is located in such a way that the yoke and antenna could be mated without any vertical movement of the antenna.



SPS-1925

# Precursor SPS Construction Base

BOEING



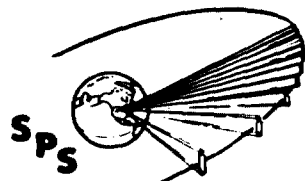
**PRECURSOR SATELLITE  
LEO BASE CREW SIZE/CONSTRUCTION TIME**

This chart shows the results of a preliminary comparison of three concepts for constructing the precursor satellite.

Alternative A is a very optimistic approach wherein it was assumed that the operational rates designed into the baseline full-size SPS construction concept could be achieved when making the precursor. The other major assumption was that the modules and antennas are constructed simultaneously. This results in a large crew but achieves a very short construction time.

Alternative B is the same as Alternative A except that a machine rate 25% as high as the operational satellite was assumed since in the early days of construction a learning process will be in effect in addition to more machine down time.

Alternative C is the concept selected as the reference approach. In this concept, it was assumed that it would be necessary to minimize the crew size. This is achieved by having a smaller number of people perform all of the construction tasks in a series approach; 1) assemble subassemblies, 2) assemble the two modules, 3) assemble the yoke, and then 4) assemble the antenna. Again, 25% of the designed operational rates were assumed. This approach will obviously take longer than the other concepts but will result in a lower capital investment cost.



SPS-1810

D180-24071-3

# Precursor Satellite

## LEO Base Crew Size/Construction Time

BOEING

### CONCEPT

#### BASELINE SYSTEM

(A)

- ANT AND MODULE CONSTRUCTED IN PARALLEL
- 100% OPERATIONAL MACHINE RATES

(B)

- ANT AND MODULE CONSTRUCTED IN PARALLEL
- 25% OPERATIONAL MACHINE RATES

(C)

- ANT AND MODULE CONSTRUCTED IN SERIES
- 25% OPERATIONAL MACHINE RATES

### CREW SIZE

|              |    |
|--------------|----|
| BASE MGMT    | 10 |
| CONSTRUCTION |    |
| MGMT         | 22 |
| MODULE CONST | 68 |
| ANT CONST    | 82 |
| SUBASSY      | 49 |
| MAINT        | 49 |
| LOGISTICS    | 42 |
| TEST/QL      | 40 |
| BASE OPS     | 39 |
| BASE SUPPORT | 77 |

10

10

6

|              |    |
|--------------|----|
| MGMT         | 22 |
| MODULE CONST | 68 |
| ANT CONST    | 82 |
| SUBASSY      | 49 |
| MAINT        | 49 |
| LOGISTICS    | 42 |
| TEST/QL      | 40 |
| BASE OPS     | 39 |
| BASE SUPPORT | 77 |

16

16

10

40

40

40

76

76

0

16

16

0

12

12

12

4

4

4

10

10

5

12

12

5

17

17

17

### TOTAL

478

213

213

99

### CONSTRUCTION TIME

340 DAYS

59 DAYS

242 DAYS

404 DAYS

- ALL OF THE NEW CONCEPTS UTILIZE IDENTICAL FACILITY

### **PRECURSOR SPS OPERATIONS SCENARIO**

The precursor program envisioned requires approximately 4-1/2 years of in-space operations. The in-space operations begin with the cargo and crew launches associated with the placement, assembly, and checkout of the construction base. These launches continue for approximately 2 years at which point additional launches begin to bring up the mass of the precursor satellite itself. The first and second bays of the satellite are assembled into one module which is dispatched, using the self-power method, to geosynchronous orbit. At the time of dispatch, assembly and checkout of the third and fourth bays are started in order to form the second module. This module carries the antenna with it to geosynchronous orbit. The self-power transfer time is 180 days. Before the first module arrives in geosynchronous orbit a manned geosynchronous orbit support station is made operable. This support station is the base of operations for the berthing operation (which joins the two modules together), final checkout and make-operable operations, and for the one year operational period baselined for this precursor satellite. It is from this support station that maintenance operations will be accomplished. Prior to the beginning of power transmission from space the test rectenna and its associated instrumentation made operable.

**• BEING**

|  | YEAR 1 | YEAR 2 | YEAR 3 | YEAR 4 | YEAR 5 |
|--|--------|--------|--------|--------|--------|
|--|--------|--------|--------|--------|--------|

(6)

**{21}**

(41)

(47)

[illegible][illegible]

## ASSEMBLE AND CHECKOUT CONST BASE

**SELF POWER  
BAYS 1 AND 2**

**ASSEMBLE  
BAYS 3 AND 4**

**SELF POWER  
BAYS 3 AND 4  
AND ANTENNA**

## ASSEMBLE ANT

▽ ▽ ▽      ▽      ▽      ▽      ▽      ▽

## MANNED GEOSYNCHRONOUS SUPPORT STATION OPERATIONS

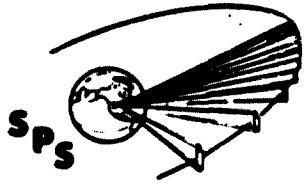
## INSTALL RECTENNA AND INSTRUMENTATION

## ONE MEGAWATT FROM SPACE

**PRECURSOR SPS MASS**

The mass is given for a precursor SPS which is 1.56% of the full system size. Because the full size SPS development will not be complete at the time of the precursor, "novelty" multiplying factors are used, with the largest factor applied to the transmitter elements (such as the klystron). To the mass of the precursor SPS in geosynchronous orbit must be added the mass of the systems necessary to accomplish self-power transfer to geosynchronous orbit, including the argon propellant, electric thrusters, power processors, etc. The total mass to be placed in low orbit for accomplishment of the precursor program is thus approximately 3600 metric tons. If a geosynchronous orbit assembly site was selected, with transfer of the satellite equipment to that orbit by chemical orbit transfer vehicles, the initial mass in low orbit would be approximately 6200 metric tons.





SPS-1821

D180-24071-3

## Precursor SPS Mass (Metric Tons)

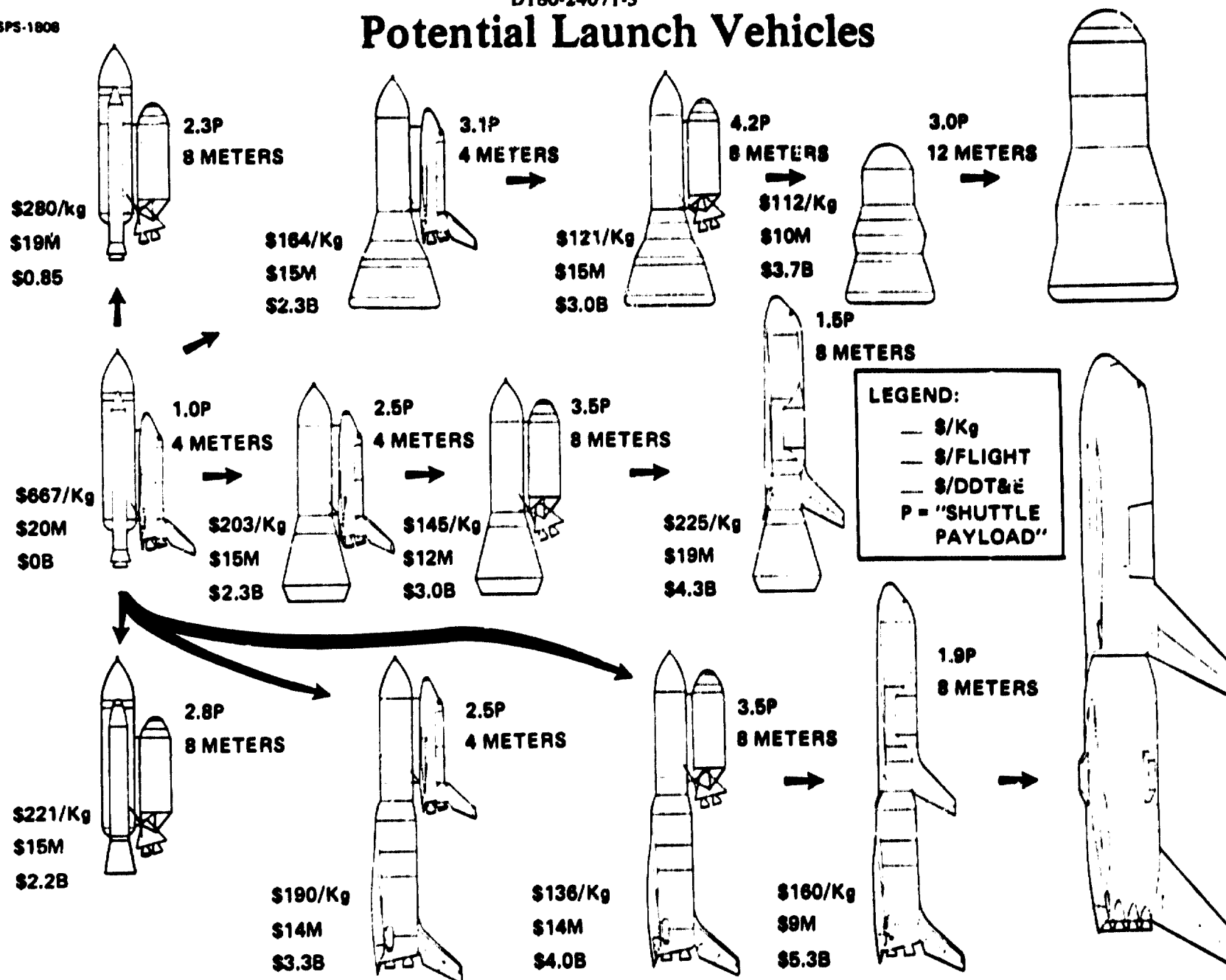
BEING

| ELEMENT                                                               | FULL SPS<br>MASS<br>(W/GROWTH) | FRACTION OF<br>FULL ELEMENT | "NOVELTY/<br>SUBSCALE"<br>FACTOR | PRECURSOR<br>MASS |
|-----------------------------------------------------------------------|--------------------------------|-----------------------------|----------------------------------|-------------------|
| POWER GENERATION                                                      | 66,000                         | 0.0156                      | 1.10                             | 1,130             |
| POWER TRANSMISSION<br>(ONE TRANSMITTER)                               | 12,600                         | 0.0312                      | 1.25                             | 490               |
| PRECURSOR SPS IN GEO ORBIT                                            |                                |                             |                                  | 1,620             |
| INITIAL MASS IN LOW ORBIT FOR SELF POWER WITH 5% SPARES (1.40 FACTOR) |                                |                             |                                  | 2,520             |
| ALLOWANCE FOR PALLETS, ETC. IN LOW ORBIT LAUNCHER (1.10) INCLUDED     |                                |                             |                                  | 2,772             |
| LOW ORBIT CONSTRUCTION BASE                                           |                                |                             |                                  | 850               |
| TOTAL MASS LAUNCHED IN PRECURSOR SPS PROGRAM                          |                                |                             |                                  | 3,620             |

### POTENTIAL LAUNCH VEHICLES

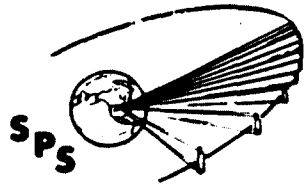
On the center left is shown the space shuttle which is nearing the end of its development phase. On the right are shown two heavy lift launch vehicle concepts which have been developed for achievement of a solar power satellite program. The upper of the two is a two-stage ballistic vehicle employing ocean landing for each of the stages. The lower is a two-staged winged vehicle which uses runway landings for recovery. Both of these have payloads in excess of 400 metric tons. They are larger than would be required for a precursor program and it is probably inappropriate to baseline their use in a precursor program. In fact one of the major advantages of a precursor as a demonstrator is that it allows successful accomplishment of a small size "power from space program" before commitment to maximum funding for either of these vehicles. The other vehicle configurations shown on the chart are derivatives of the space shuttle. Above the space shuttle is shown a vehicle employing an 8 meter diameter payload shroud with a SSME recovery capsule. This vehicle uses the standard external tank and solid rocket boosters. The payload is approximately 2.3 times that of the standard shuttle. Below it is a similar vehicle which instead of the solid rocket boosters employs liquid rocket stages which are recovered at sea. The space shuttle orbiter and the 8-meter diameter payload shroud/SSME combination are also shown with two types of ballistically recovered liquid rocket stages, and with a winged turbojet flyback booster. The larger of the ocean-recovery ballistic rocket stages is also shown with a ballistic orbiter; the smaller is shown with an internally tanked liquid hydrogen/liquid oxygen orbiter vehicle employing SSME's. This internally tanked orbiter is also shown in combination with the winged booster. For each of the shuttle derivative vehicles the multiple of the shuttle payload, the diameter of the payload, the cost per kilogram to low orbit, the cost per flight, and the total development of that vehicle are given. Each of these development costs assumes that successful completion of the entire shuttle development program has occurred, and that the full SSME, reusable insulation, etc. is available.

# Potential Launch Vehicles



**LOW ORBIT PLACEMENT COSTS**

This chart shows the total cost required to place an accumulated mass figure into low orbit. The costs include not only the per flight cost but also the developmental cost association with that vehicle. The mass required to accomplish the 1.56% size precursor program is shown along with a mass figure for that precursor and the mass accumulated by 15 years of vehicle operation at a launch rate of 40 flights per year. The line associated with the space shuttle begins at 0 since it is assumed here that its development has been accomplished. However, its relatively high per-flight cost causes it to be above the other systems before eight years of operation have taken place. The other shuttle derivatives have higher DDT&E cost but lower cost per flight. The lowest cost vehicle for placement of the precursor SPS shown here is the shuttle derivative employing the 8 meter diameter shroud and a SSME recovery capsule with the standard external tank and solid rocket boosters.

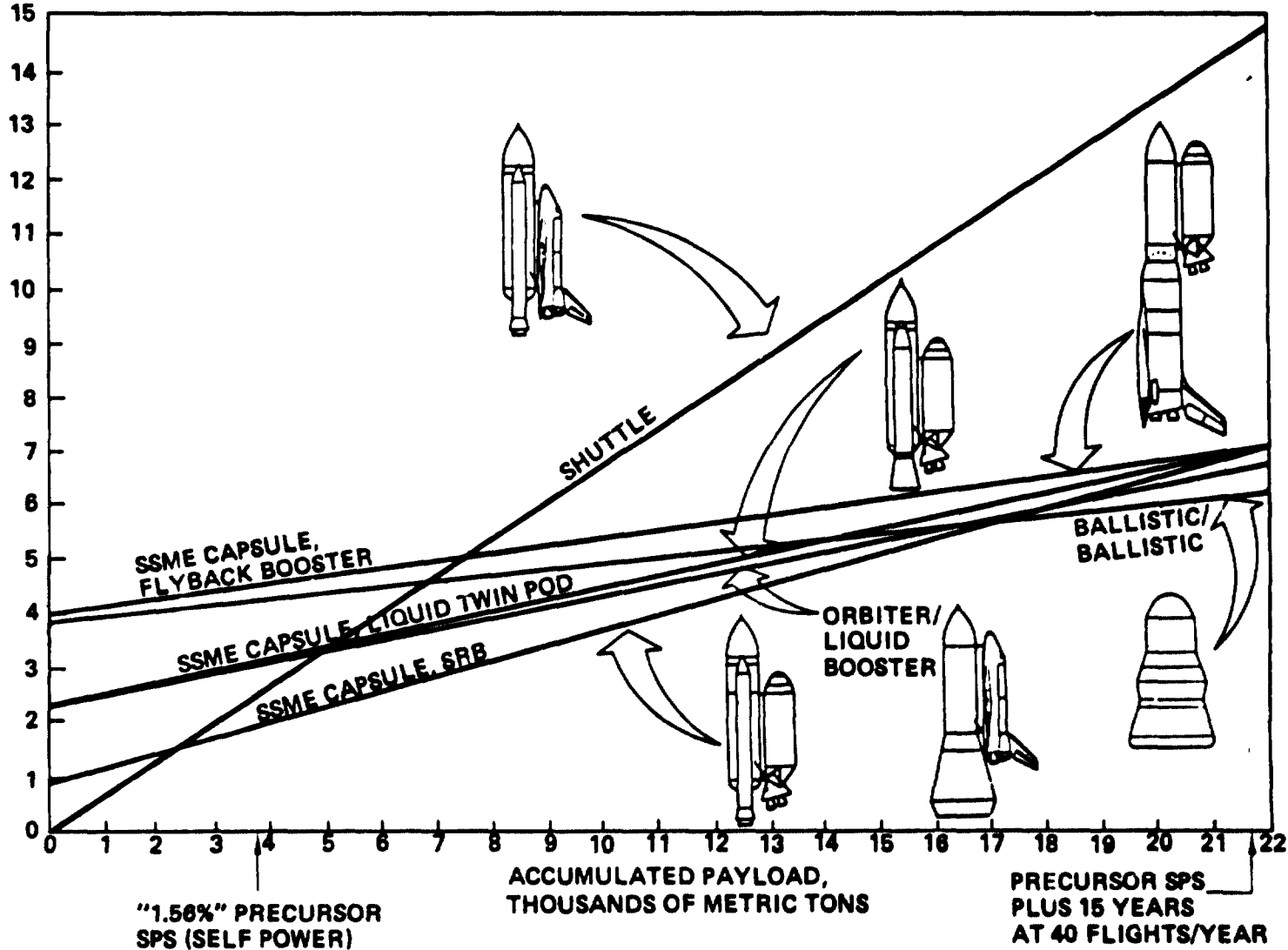


# Low Orbit Placement Costs (No Discounting)

BOEING

SPS-1853

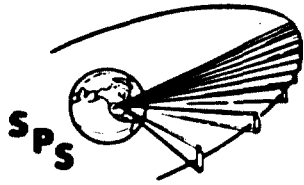
LAUNCH  
VEHICLE  
DDT&E +  
LAUNCH  
COSTS



**THE SRB AND STRATOSPHERIC OZONE**

This sentence is an excerpt from the draft environmental impact statement for the space shuttle orbiter and is indicative of the environmental concerns associated with large scale operations of the solid rocket boosters for an extended period of time.

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SPS-1825

## The SRB and Stratospheric Ozone

**BOEING**

"THE DECAY TIME OF THE SPACE SHUTTLE EXHAUST EFFECT IS CORRESPONDINGLY SHORT, SO THAT AFTER REPLACEMENT OF THE CURRENT BOOSTER BY A NONCHLORINE BOOSTER, THE OZONE LAYER WOULD RETURN TO NORMAL IN A FEW YEARS."

ENVIRONMENTAL IMPACT STATEMENT, SPACE SHUTTLE PROGRAM

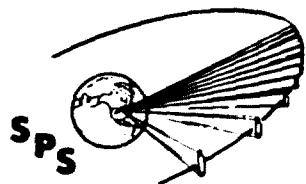
(DRAFT)

JULY, 1977

**LOW VERSUS HIGH ORBIT ASSEMBLY FOR PRECURSOR SPS**

In this chart it is assumed that 1/4 of the development cost associated with shuttle derivative launch vehicles is chargeable to the precursor SPS program. Costs for the transportation elements of low and high orbit assembly operations are given for the shuttle and two shuttle derivative vehicles. With the shuttle, low orbit assembly is far lower in cost than with high orbit. With one derivative, low orbit assembly is slightly more expensive. We may conclude that self-power assembly for the precursor program will approximately pay for itself and that therefore it should be assumed to be part of the program, since it demonstrates and develops the eventual self-power transfer system.





D180-24071-3

## Low Versus High Orbit Assembly for Precursor SPS

SPS-1856

BOEING

(WITH ¼ OF LAUNCH VEHICLE DDT&E) (\$B, 1977)

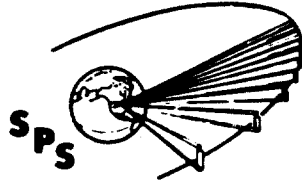
|                               | SHUTTLE     | SRB/ET<br>SSME CAPSULE | FLYBACK/ET<br>SSME CAPSULE |
|-------------------------------|-------------|------------------------|----------------------------|
| <b>LOW ORBIT (SELF POWER)</b> |             |                        |                            |
| LAUNCH 3,790 MT               | 2.52        | 1.27                   | 1.52                       |
| DEVELOP SELF POWER            | 0.60        | 0.60                   | 0.60                       |
| SELF POWER EQUIPMENT          | 0.25        | 0.25                   | 0.25                       |
| DEV 40 MT OTV                 | 0.20        | 0.20                   | 0.20                       |
| NINE OTV FLIGHTS              | 0.04        | 0.04                   | 0.04                       |
|                               | <u>3.61</u> | <u>2.36</u>            | <u>2.61</u>                |
| <b>HIGH ORBIT</b>             |             |                        |                            |
| LAUNCH 6,240 MT               | 4.16        | 1.96                   | 1.66                       |
| DEV 40 MT OTV                 | 0.20        | 0.20                   | 0.20                       |
| 51 OTV FLIGHTS                | 0.21        | 0.21                   | 0.21                       |
|                               | <u>4.57</u> | <u>2.37</u>            | <u>2.26</u>                |

CONCLUSION: SELF POWER ABOUT PAYS FOR ITSELF AND DEMONSTRATES/DEVELOPS FINAL SYSTEM.

**D180-24071-3**

**TOTAL COST THROUGH NUMBER 1 SPS PHOTOVOLTAIC SYSTEM**

This chart shows development, facilitization and unit costs for the SPS program through the first full size unit. Note that the development cost for the SPS itself are relatively small in comparison with the total. This indicates that a good precursor program may accomplish nearly all of the direct development associated with the SPS. This will be shown on the next chart.

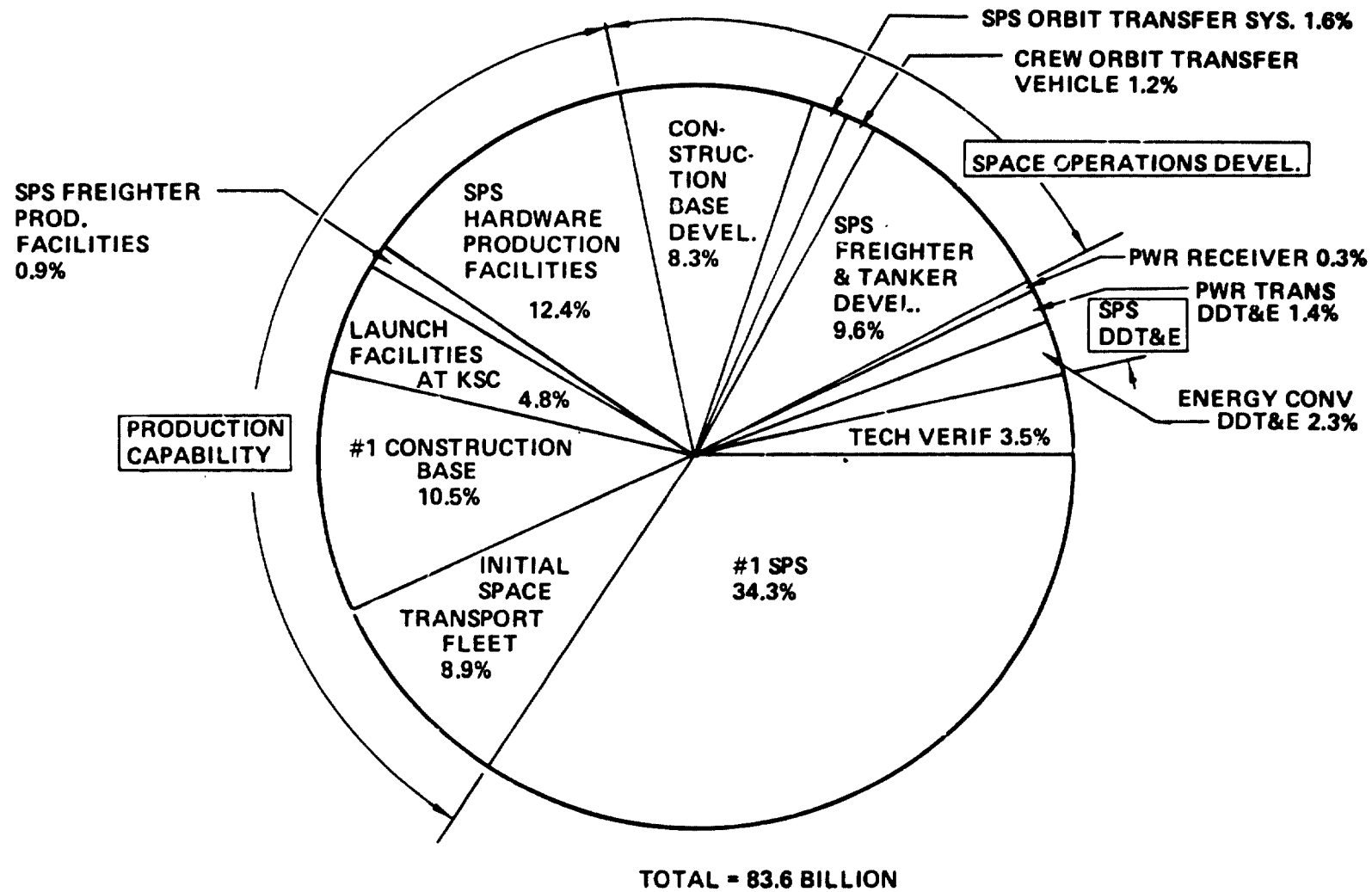


D180-24071-3

## Total Costs Through #1 SPS Photovoltaic System

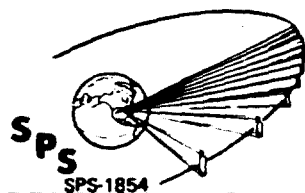
SPS-1549

BOEING



**PRECURSOR SPS COST ESTIMATE**

A preliminary cost estimate is made for each of the major elements of the precursor program, along with a 15% allowance for miscellaneous items, operational costs, etc. which have not been identified. Note that not all the cost associated with the shuttle derivative launch vehicle, its facilities, the chemical OTV, etc. are directly charged to this program, since it is quite probable that all of these vehicles will have other uses. In fact it is probable that one would not embark upon a shuttle derivative launch vehicle unless it had significant other use than for a precursor SPS program. Many of the cost elements have direct applicability to the full size SPS so that development funds expended in the precursor program directly reduces the funding necessary to accomplish the eventual SPS. The total value of this reduction is approximately \$6 billion for the precursor program.



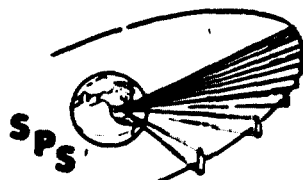
# Precursor SPS Cost Estimate ("1.56%")

BOEING

| ELEMENT                                                   |                          | \$B (1977) | APPROXIMATE CONTRIBUTION<br>TO SPS DDT&E \$B (1977) |
|-----------------------------------------------------------|--------------------------|------------|-----------------------------------------------------|
| CONSTRUCTION BASE (WITH \$3.0B DDT&E)                     |                          | 5.30       | 3.1                                                 |
| SPS DDT&E:                                                | POWER GENERATION         | 0.96       | 0.8                                                 |
|                                                           | POWER TRANSMISSION       | 0.59       | 0.4                                                 |
|                                                           | POWER RECEPTION          | 0.12       | 0.1                                                 |
| SPS HARDWARE:                                             | POWER GENERATION         | 0.35       |                                                     |
|                                                           | POWER TRANSMISSION       | 0.25       |                                                     |
|                                                           | STRUCTURE, MISCELLANEOUS | 0.20       |                                                     |
| SELF POWER TRANSFER (WITH DDT&E)                          |                          | 0.85       | 0.6                                                 |
| GSO SUPPORT STATION (WITH DDT&E)                          |                          | 1.20       | 0.4                                                 |
| LEO TRANSPORT (FLYBACK BOOSTER/ET/8M SHROUD/SSME CAPSULE) |                          |            |                                                     |
| 1/4 OF DDT&E                                              |                          | 1.00       | 0.2                                                 |
| 47 FLIGHTS (9 SUPPORT GSO STATION)                        |                          | 0.66       |                                                     |
| FLEET (1/2 BOOSTER, 1/2 SSME CAPSULE)                     |                          | 0.80       |                                                     |
| FACILITIES (1/4 PAD, PAYLOAD HANDLING, ETC.)              |                          | 0.40       |                                                     |
| CHEMICAL OTV (40 MT CLASS, 1/4 DDT&E)                     |                          | 0.40       | 0.3                                                 |
| CREW ROTATION (75 PERSON CARRIER)                         |                          |            |                                                     |
| DDT&E                                                     |                          | 0.16       | 0.16                                                |
| 25 SHUTTLE LAUNCHES (OVER 3 YEARS)                        |                          | 0.50       |                                                     |
| RECTENNA (ONE MEGAWATT OUT)                               |                          | 0.70       |                                                     |
| SUBTOTAL                                                  |                          | 14.43      | 6.06                                                |
| WITH 15% FOR OPERATIONS, MISCELLANEOUS                    |                          | 16.57      |                                                     |

**SIZE EFFECT ON PRECURSOR PROGRAM COST**

Three sizes for precursor SPS were investigated in this study. The 2-bay unit is 0.78% of the full size SPS. The 4-bay unit baselined is 1.56% of the full size SPS, and a 32-bay (one module) unit which is 12.5% of the full size SPS. Because it is considered fundamental that one of the major functions of the precursor unit is to develop and demonstrate the construction base "throughput" required for commercial viability of the SPS, each of these precursor unit sizes has associated with it a 1/8 size construction base segment which is literally a segment of the full size construction base. Because of this and other scaling effects there is relatively little cost difference between the 2-bay and 4-bay unit. If it is further baselined that 1 megawatt of useful power is to be produced by the test rectenna the smaller unit actually has a higher total program cost. Because of the lower central beam strength of the smaller precursor (approximately 5 microwatts per square centimeter) it requires a much larger test rectenna to generate the 1 megawatt. The one module large precursor SPS requires only a very small rectenna to generate one megawatt.

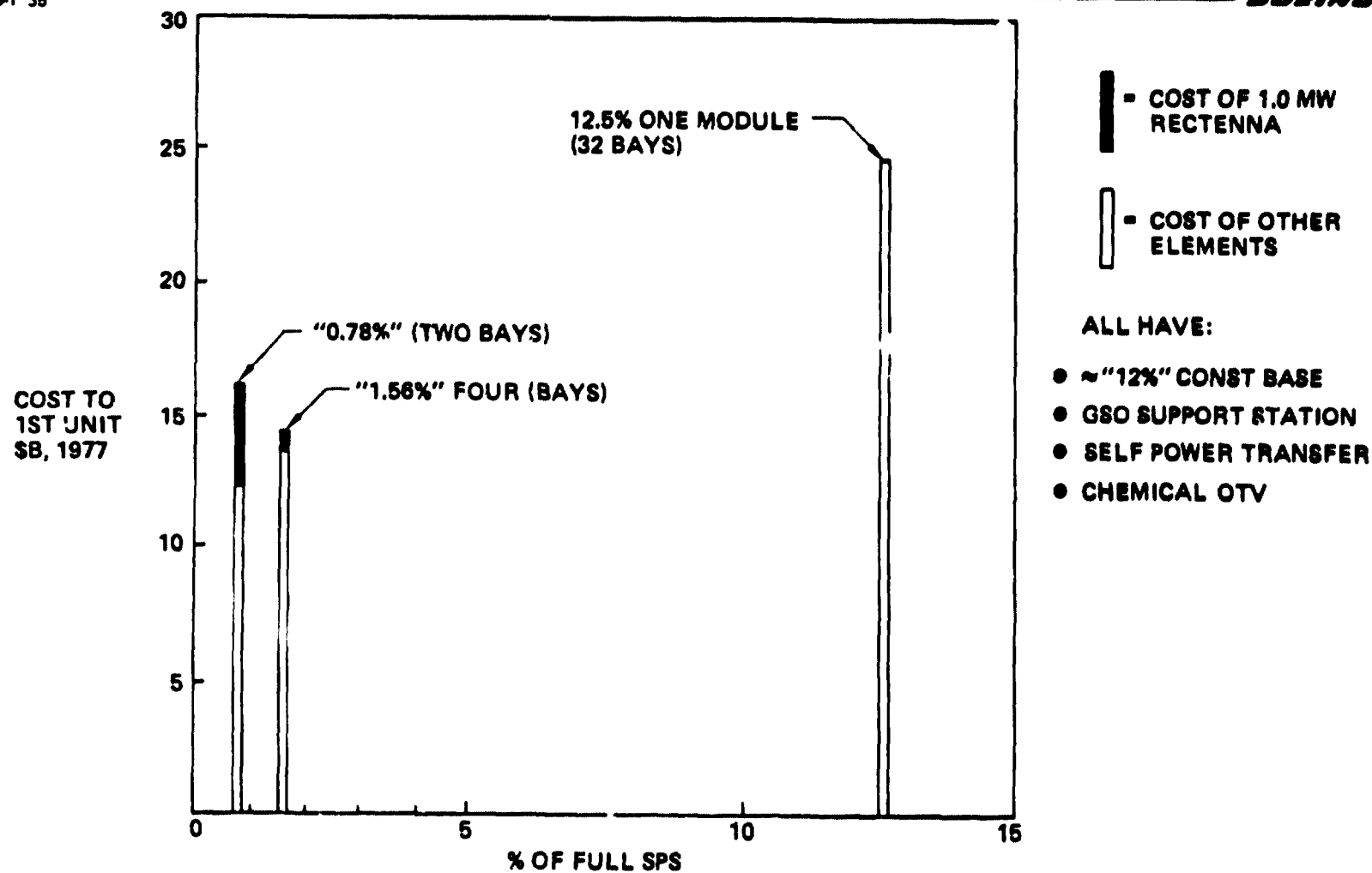


SPS-1' 35

D180-24071-3

## Size Effect on Precursor Program Cost

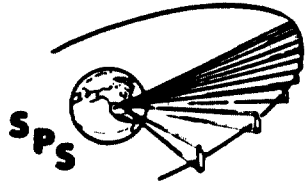
BOEING



### **OTHER PRECURSOR SIZE OPTIONS**

This chart shows the major effects of selecting the smaller and larger precursor units investigated. The 12.5% unit is one complete SPS module and would allow duplication of this large SPS assembly, certainly a major contributor to demonstration of the full satellite construction capability. However, its larger size and mass will require a much more ambitious associated launch program. For example, if the shuttle were to be used exclusively, over 800 shuttle flights would be required, and even with the higher payload capability of the shuttle derivative launch vehicle, over 200 flights are required. Total precursor program cost is estimated at approximately \$28 billion. This larger size unit allows a central beam strength of over 1 milliwatt per square centimeter, much stronger than that producible by the 4-bay unit. Consequently, a very small rectenna would be sufficient to produce the 1 megawatt output baselined. Conversely, the 0.78%, 2-bay, precursor would allow a much more modest launch program and might be accomplished by the shuttle alone. However, the full high voltage output of the solar arrays would be achievable only with some cell string configuration other than that baselined for the full size SPS. The lower central beam strength requires a much larger rectenna if the full 1 megawatt is to be produced.





SPS-1857

## Other Precursor Size Options

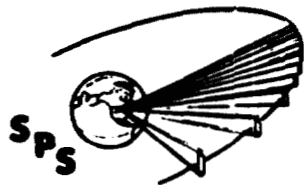
BOEING

- **"12.5%": ONE SPS MODULE (32 BAYS)**
  - EXACT DUPLICATION OF EVENTUAL SPS CONSTRUCTION ITEM.
  - 810 SHUTTLE FLIGHTS OR 225 FLIGHTS OF LIQUID BOOSTER/CAPSULE SSME SHUTTLE DERIVATIVE
  - APPROXIMATELY \$28B PROGRAM.
  - $1.37 \text{ mW/cm}^2$  ( $1,370 \text{ } \mu\text{W/cm}^2$ ) CENTRAL BEAM STRENGTH.
  - \$50M FOR A 1.0 mW ANTENNA
- **0.78%: TWO SPS BAYS**
  - CONSTRUCT ALL SPS "ELEMENTS".
  - FULL VOLTAGE ONLY BY EITHER SUBSCALE CELLS OR RE-ROUTED STRINGS. (NOT DUPLICATION OF PLASMA SUSCEPTIBILITY).
  - 95 SHUTTLE FLIGHTS OR 31 FLIGHTS OF LIQUID BOOSTER/CAPSULE SSME SHUTTLE DERIVATIVE.
  - $1.40 \text{ } \mu\text{W/cm}^2$  CENTRAL BEAM STRENGTH
  - \$3.8B FOR A 1.0 mW ANTENNA.

### **PRECURSER GOALS**

The goals that are thus emerging from the study may be summarized as follows. The precursor will operate in geosynchronous orbit and develop for a test period (probably one year) approximately 1 megawatt of useful power; while doing so, an environmentally acceptable microwave beam will be continuously and accurately pointed into the required receiving area. By accomplishing this test, all major elements of the full size SPS will be demonstrated, including the high required construction "throughput" necessary to assemble satellites on orbit at the required rate. The precursor satellite is relatively small compared to a full size SPS and, consequently, is capable of being launched along with its construction base by the space shuttle or a derivative of the space shuttle.

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SPS-1880

D180-24071-3

## Precursor Goals

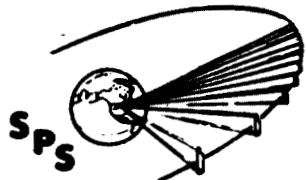
BOEING

- **ALLOWS LONG-TERM (APPROXIMATELY 1 YEAR) DEMONSTRATION THAT MICROWAVE BEAM CAN BE CONTINUOUSLY POINTED INTO REQUIRED AREA**
- **BEAM TO FIT INTO GOVERNMENT RESERVATION (e.g., WHITE SANDS PROVING GROUND); SIGNAL STRENGTH AT EDGE TO BE BELOW SOVIET EXPOSURE STANDARD ( $10 \mu\text{W}/\text{cm}^2$ )**
- **CENTRAL BEAM STRENGTH TO ALLOW WORKERS EXPOSURE (UNDER 1/10 CURRENT U.S. STANDARD)**
- **SIGNIFICANT POWER OUTPUT ( $> 1 \text{ MW}$ )**
- **TO REPRESENT ELEMENTS OF FULL-SIZE SPS**
  - **POWER GENERATION**
  - **TRANSMITTER**
  - **ROTARY JOINT**
  - **TRANSFER TO HIGH ORBIT**
  - **CONSTRUCTION "THROUGHPUT"**
  - **POWER RECEPTION**
  - **MAINTENANCE**
- **SHUTTLE DERIVATIVE ADEQUATE FOR ALL LAUNCHES**

**MAJOR ELEMENTS OF THE PRECURSOR PROGRAM**

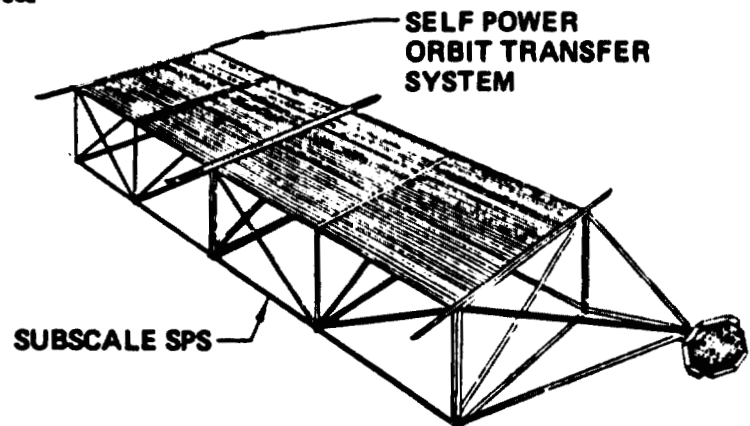
These are the larger elements of the 1.56% precursor program baselined in this study. In addition to the precursor satellite itself with its transmitter, the self-power orbit transfer system is required. A low earth orbit assembly and checkout facility (construction base) is also a major element of the program. Some shuttle derivative launch vehicle, preferably with an eight meter diameter cargo capability, should be provided in order to reduce launch cost not only for the precursor program but also subsequent shuttle type operations. A chemical orbit transfer vehicle is required to place the manned geosynchronous orbit support station which is used during the operational test period of the precursor satellite.

# Major Elements of the Precursor Program



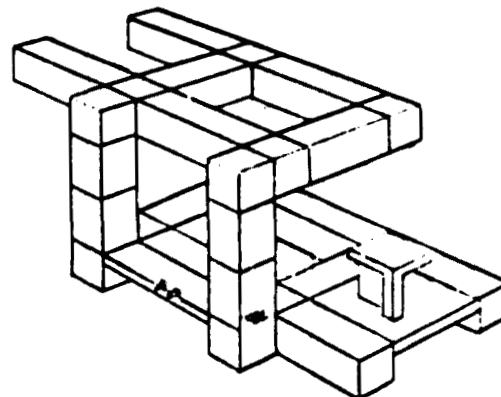
SPS-1852

**BOEING**

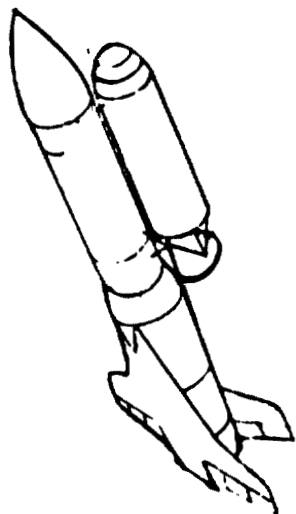


**SELF POWER  
ORBIT TRANSFER  
SYSTEM**

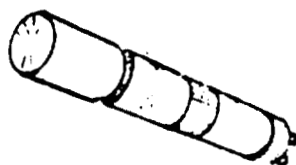
**SUBSCALE SPS**



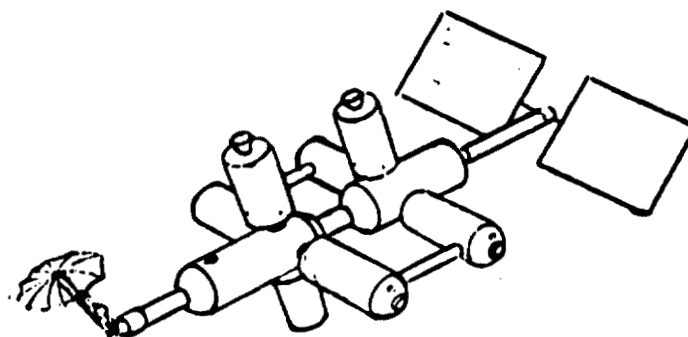
**LEO ASSEMBLY AND  
CHECKOUT FACILITY**



**SHUTTLE DERIVATIVE  
LAUNCH VEHICLE**



**CHEMICAL  
ORBIT TRANSFER  
VEHICLE**



**GEOSYNCHRONOUS  
SUPPORT STATION**

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PHASE III FINAL REVIEW

MARCH 1, 2, 1978

# **SOLAR POWER SATELLITE**

**GENERAL  ELECTRIC**  
**SPACE DIVISION**

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MICROWAVE POWER TRANSMISSION

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SCOPE OF GE'S PHASE CONTROL SYSTEM STUDY

GENERAL ELECTRIC COMPANY'S STUDY EFFORT IN THE PHASE CONTROL CIRCUIT AREA WAS CONCENTRATED ON THE DEVELOPMENT OF SYSTEM ARCHITECTURE, DEFINITION OF MAIN REQUIREMENTS AND ANALYSIS OF THE POWER TRANSFER EFFICIENCY OF THE SYSTEM.





SCOPE OF GE'S PHASE CONTROL  
SYSTEM STUDY

- SURVEY AND INTEGRATE PRIOR EFFORTS
- COMPARE ALTERNATE APPROACHES
- DEFINE MAIN SYSTEM PARAMETERS ON THE BASIS OF TRADE-OFF ANALYSIS
- ANALYZE OPERATION OF SYSTEM AND DETERMINE ERRORS CAUSING POWER TRANSFER INEFFICIENCY.
- PROVIDE DATA FOR COST MODEL.

BASIC ASSUMPTIONS FOR SYSTEM DESIGN OF PHASE CONTROL NETWORK

REALISTIC AND RELATIVELY EASILY MAINTAINABLE ORBIT PARAMETERS WERE ASSUMED. ORBIT INCLINATION ANGLE LIMIT NECESSITATES A DAILY MECHANICAL OR ELECTRICAL TILT ALIGNMENT OF SUBARRAY PATTERN.



BASIC ASSUMPTIONS FOR SYSTEM DESIGN OF  
PHASE CONTROL NETWORK

ORBIT

|                          |                    |
|--------------------------|--------------------|
| EXCENTRICITY             | $4 \times 10^{-4}$ |
| INCLINATION              | $2.2^{\circ}$      |
| LONG TERM POSITION DRIFT | $\pm 10$ km        |

BASIC ASSUMPTIONS FOR SYSTEM DESIGN OF PHASE CONTROL NETWORK

THE GOVERNING CHARACTERISTICS FOR THE SPACE SEGMENT OF THE PHASE CONTROL SYSTEM WERE SELECTED ON A CONSERVATIVE BASIS. WITH EVOLVING CONSTRUCTION TECHNOLOGY OF THE ANTENNA STRUCTURE THE NUMBER OF SUBARRAYS MAY BE REDUCED RESULTING IN A SIGNIFICANT REDUCTION OF PHASE CONTROL SYSTEM COMPLEXITY.



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**BASIC ASSUMPTIONS FOR SYSTEM DESIGN OF  
PHASE CONTROL NETWORK**

**SPACE SEGMENT**

|                                                                     |                                    |
|---------------------------------------------------------------------|------------------------------------|
| NOMINAL SPACECRAFT ANTENNA DIAMETER                                 | 1 km                               |
| NUMBER OF SUBARRAYS                                                 | 10000                              |
| NOMINAL SUBARRAY AREA                                               |                                    |
| TRANSMIT                                                            | $10^2 \text{ m}^2$                 |
| RECEIVE                                                             | $5.33^2 \text{ m}^2 \text{ (max)}$ |
| ARRAY MISALIGNMENT                                                  | $.15^\circ \text{ (max)}$          |
| POLARIZATION                                                        | LINEAR                             |
| NOMINAL TRANSMIT POWER OF ARRAY                                     | $10^{10} \text{ w}$                |
| MAXIMUM TRANSMIT POWER PER TUBE                                     | 125 kw                             |
| MAXIMUM NOISE DENSITY FOR 125 KW TUBE<br>90 MHz FROM CENTER OF BAND | -107.8 dBw/Hz                      |
| RECEIVER IF BANDWIDTH                                               | 1 MHz                              |
| PHASE DISTRIBUTION NETWORK ARCHITECTURE                             | 3 LAYERS                           |

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BASIC ASSUMPTIONS FOR SYSTEM DESIGN OF PHASE CONTROL NETWORK

A TWO TONE PILOT SIGNAL TRANSMITTED BY A THREE ELEMENT EARTH STATION ANTENNA ARRAY WAS SELECTED ALLOWING A SIMPLIFICATION OF THE SPACE SEGMENT AND COMPENSATION OF SYSTEMATIC POINTING ERRORS IN THE POWER BEAM.



BASIC ASSUMPTIONS FOR SYSTEM DESIGN OF  
PHASE CONTROL NETWORK

EARTH SEGMENT

|                                       |                       |
|---------------------------------------|-----------------------|
| EARTH STATION SITE                    | WITHIN CONTINENTAL US |
| UPLINK FREQUENCY ( $f_U$ )            | 2460 MHz              |
| DOWNLINK FREQUENCY ( $f_D$ )          | 2450 MHz              |
| UPLINK MODULATION FREQUENCY ( $f_1$ ) | 100 MHz               |
| NUMBER OF PILOT ANTENNAS              | 3                     |
| NOMINAL RECTENNA DIAMETER             | 10 km                 |
| NUMBER OF MONITORING ANTENNAS         | 4 (MIN)               |

FUNCTIONS IN PHASE CONTROL CIRCUIT FOR RETRODIRECTIVE SPS ANTENNA

THE NECESSARY FUNCTIONS REQUIRED FOR A RETRODIRECTIVE PHASE CONTROL SYSTEM WERE IDENTIFIED. THESE FUNCTIONS ARE INDEPENDENT OF THE ACTUAL CIRCUIT IMPLEMENTATION.

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FUNCTIONS IN PHASE CONTROL CIRCUIT FOR  
RETRODIRECTIVE SPS ANTENNA

- PILOT SIGNAL GENERATION, CONTROL AND TRANSMISSION AT  $f_U$  FROM EARTH STATION.
- PILOT PHASE RECEPTION AND DOWN CONVERSION AT SPACECRAFT SUB-SUBARRAYS.
- REFERENCE PHASE GENERATION, DISTRIBUTION AND REGENERATION.
- PILOT PHASE TRANSMISSION FOR PHASE REGENERATION AND CONJUGATION.
- CONJUGATED PHASE TRANSMISSION AND UP CONVERSION TO  $f_D$  FOR PA SYSTEM.
- PHASE CONTROLLED POWER AMPLIFICATION AND TRANSMISSION BY SPACECRAFT SUBARRAYS.
- MONITORING OF RECEIVED SIGNAL ON GROUND.

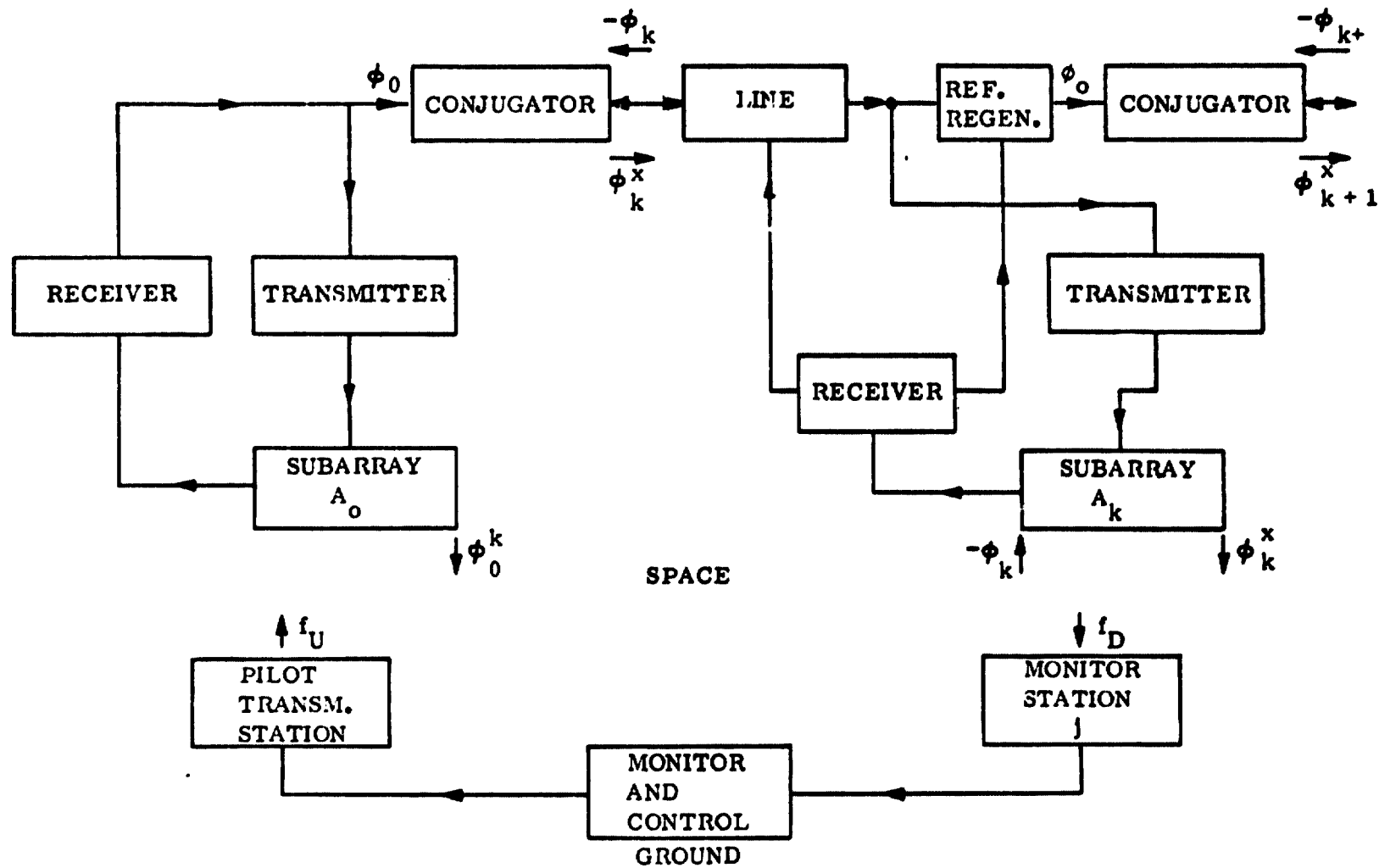
RETRODIRECTIVE SPS PHASE CONTROL SYSTEM

THE RETRODIRECTIVE PHASE CONTROL SYSTEM IS BASED ON THE USE OF A GROUND GENERATED REFERENCE PHASE AGAINST WHICH THE PHASE OF THE RECEIVED SIGNALS OF THE SUBARRAYS ARE CONJUGATED FOR THE TRANSMITTED SIGNALS. THE REFERENCE PHASE IS DISTRIBUTED ON A RETURNABLE TIME BASIS, WHICH IS INDEPENDENT ON THE VARIATIONS OF ELECTRICAL LENGTH IN THE PHASE DISTRIBUTION TRANSMISSION LINES. THE PRACTICAL IMPLEMENTATION OF THIS PRINCIPLE REQUIRES THAT THE CONJUGATOR FOR A GIVEN SUBARRAY IS LOCATED AT THE RECEIVER OF THE NEXT HIGHER LEVEL SUBARRAY IN THE PHASE DISTRIBUTION TREE.

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## RETRODIRECTIVE SPS PHASE CONTROL SYSTEM



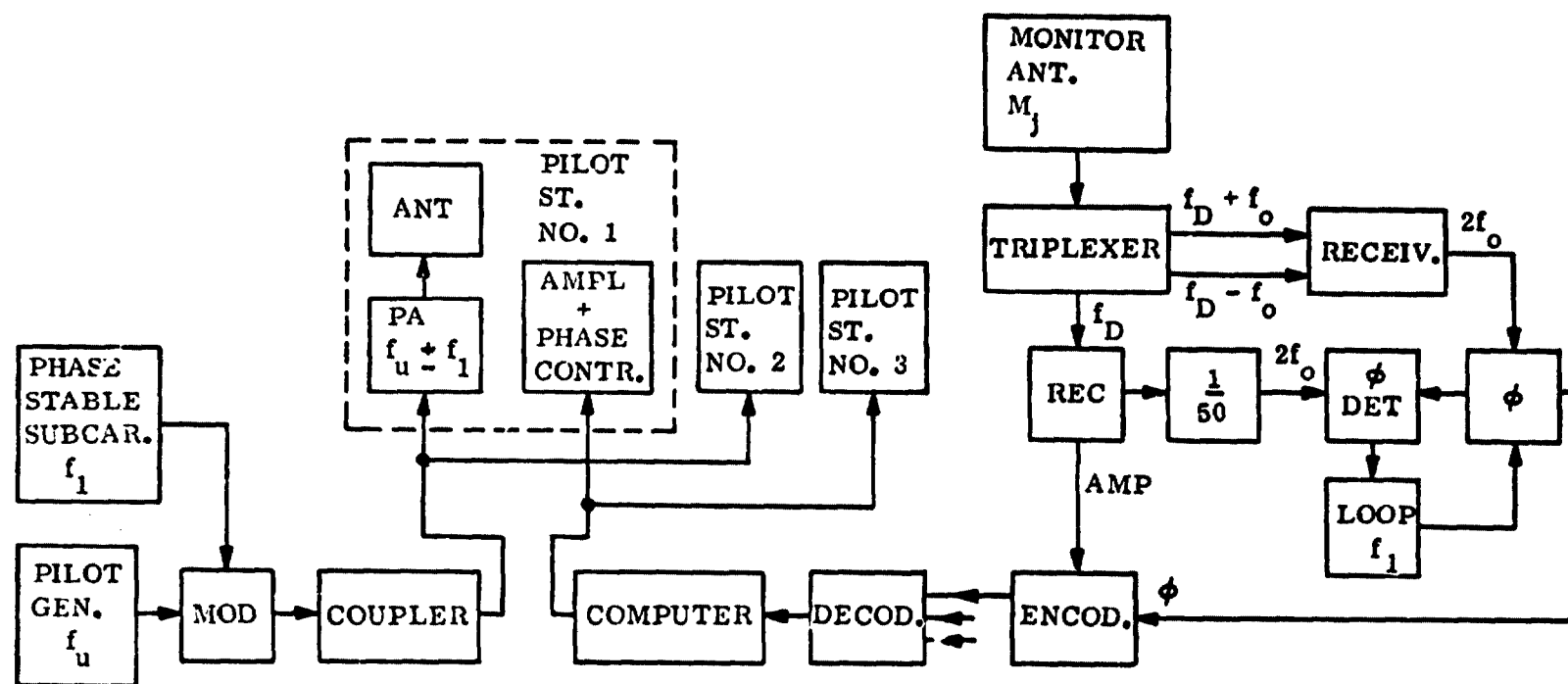
THREE PILOT ANTENNA CONTROL SYSTEM

ON THE GROUND THE POSITION OF THE RECEIVED BEAM IS MONITORED AND THE EFFECTIVE PHASE CENTER OF THE TRIANGULARLY CONFIGURED PILOT ANTENNA ARRAY IS VARIED IN SUCH A WAY THAT THE BEAM CENTER IS KEPT AT THE CENTER OF THE RECTENNA.

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## THREE PILOT ANTENNA CONTROL SYSTEM



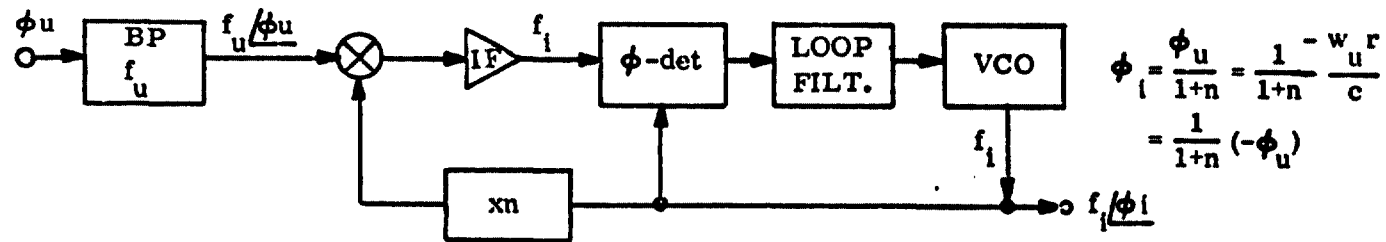
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PHASE COHERENT RECEIVER ALTERNATIVE

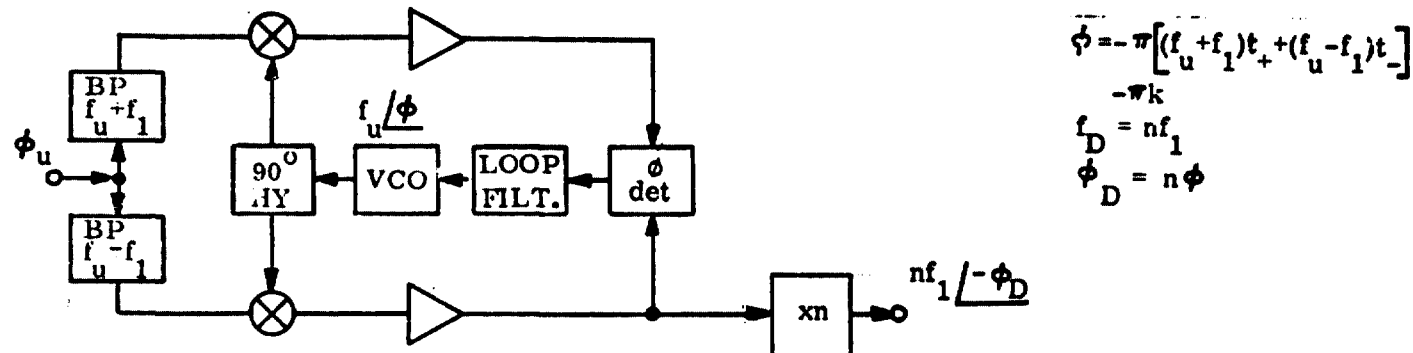
SEVERAL PHASE COHERENT RECEIVER TYPES ARE USABLE, AMONG WHICH THE TWO TONE RECEIVER WITH A FIXED LO IS THE SIMPLEST AND PRODUCES THE RECEIVED PHASE AT THE SUBARRAY AT A CONVENIENTLY LOW INTERMEDIATE FREQUENCY.



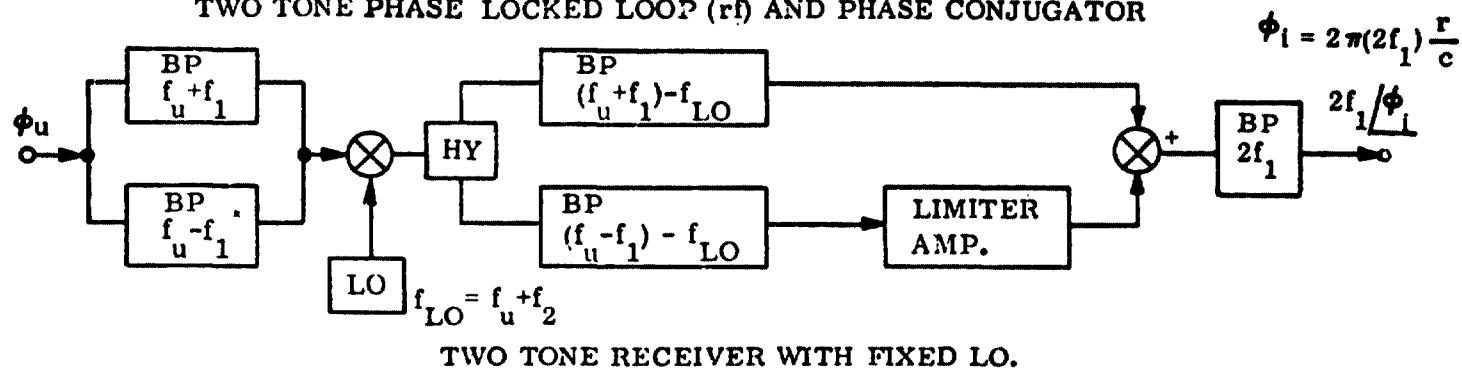
# PHASE COHERENT RECEIVER ALTERNATIVE



SINGLE TONE PHASE LOCKED LOOP (1f)



TWO TONE PHASE LOCKED LOOP (rf) AND PHASE CONJUGATOR



TWO TONE RECEIVER WITH FIXED LO.

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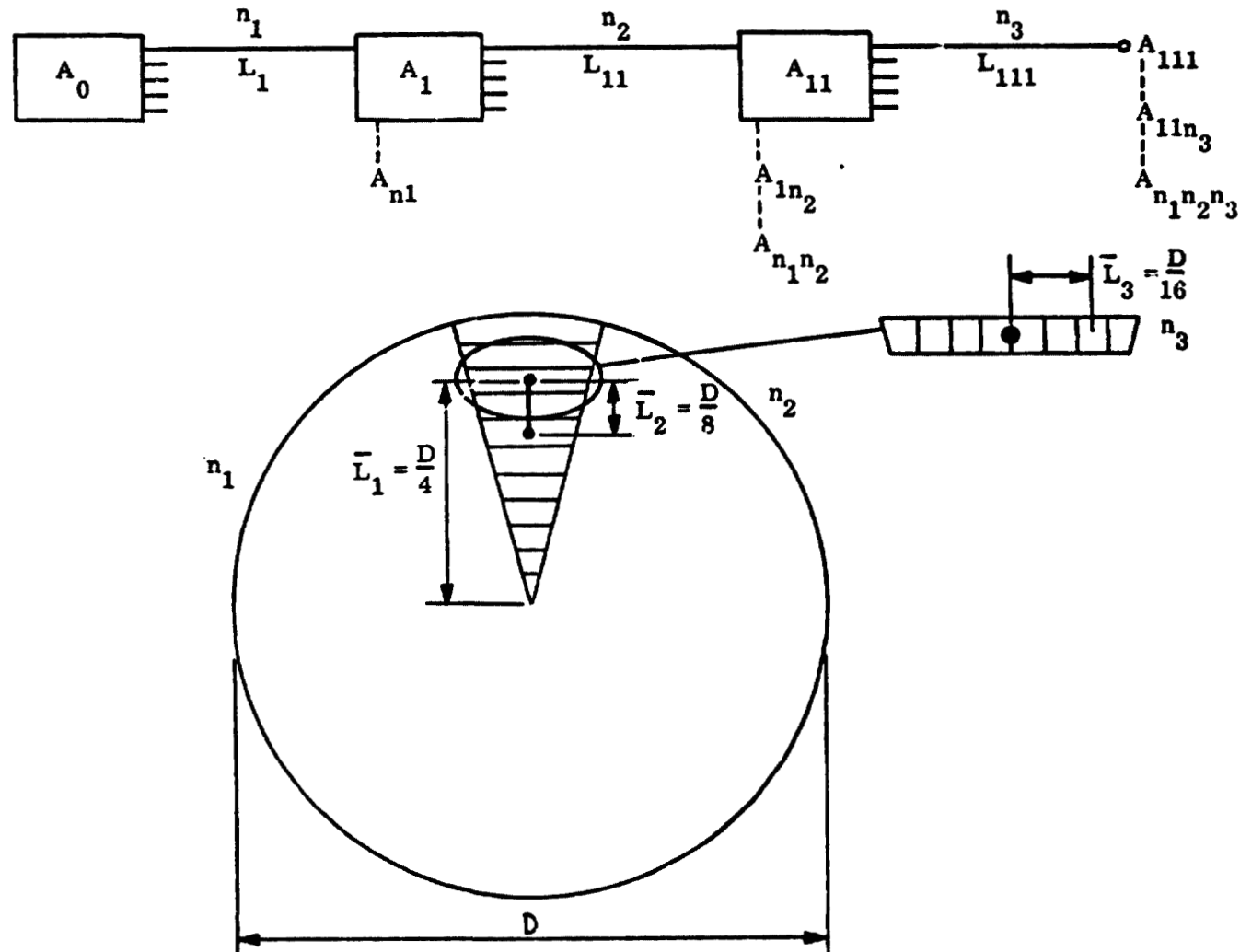
PHASE DISTRIBUTION NETWORK LAYOUT

A THREE LAYER  $i$ - $f$  REFERENCE PHASE DISTRIBUTION SYSTEM WAS SELECTED TO MINIMIZE CIRCUIT LOSSES, WEIGHT AND PRACTICAL ERRORS ASSOCIATED WITH THE TERMINATING IMPEDANCE VARIATIONS. THIS SYSTEM REQUIRES ONLY  $n_1(n_2 + 1)$  PHASE REGENERATORS ( $n_1 = 19$ ,  $n_2 = 23$ ) AND AN  $i$ - $f$  DIPLEXER AT THE END OF EACH TRANSMISSION LINES, WHICH ARE USED AT  $2f_1$  AND  $4f_1$  FREQUENCIES FOR THE BACK AND FORTH TIME RETURNING OF THE REFERENCE SIGNALS.





# PHASE DISTRIBUTION NETWORK LAYOUT



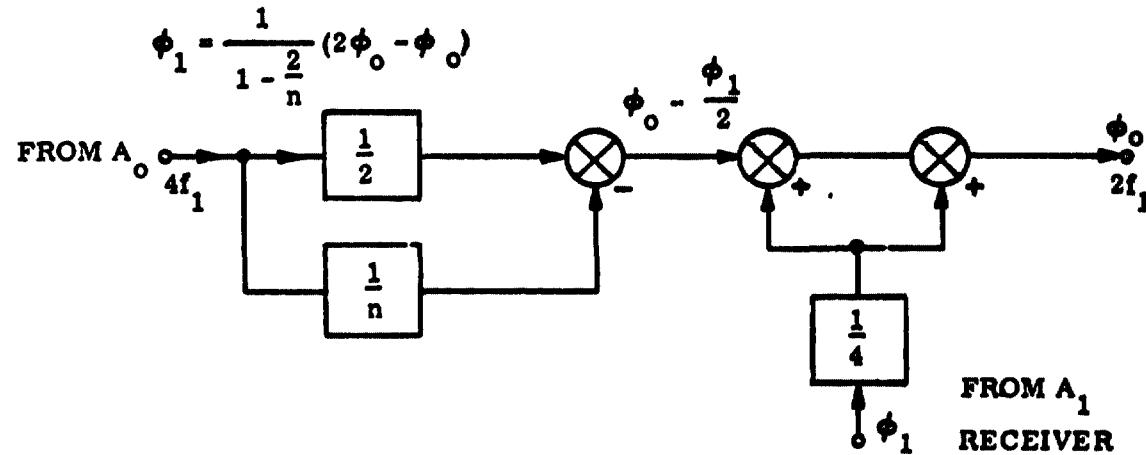
ALTERNATIVE PHASE REGENERATORS

IN THE REFERENCE PHASE DISTRIBUTION PROCESS A REGENERATION OF THE ORIGINAL REFERENCE PHASE IS REQUIRED AT EVERY NODE FROM THE RECEIVED SUBARRAY SIGNAL AND FROM THE CONJUGATED SUBARRAY SIGNAL. THIS "REGENEATOR" REQUIRES A NUMBER OF MIXERS AND FREQUENCY DIVIDERS. WHEN THE DIVIDERS ARE USED IN A LARGER OVERALL SYSTEM CARE MUST BE TAKEN BY THEIR PROPER SYNCHRONIZATION.

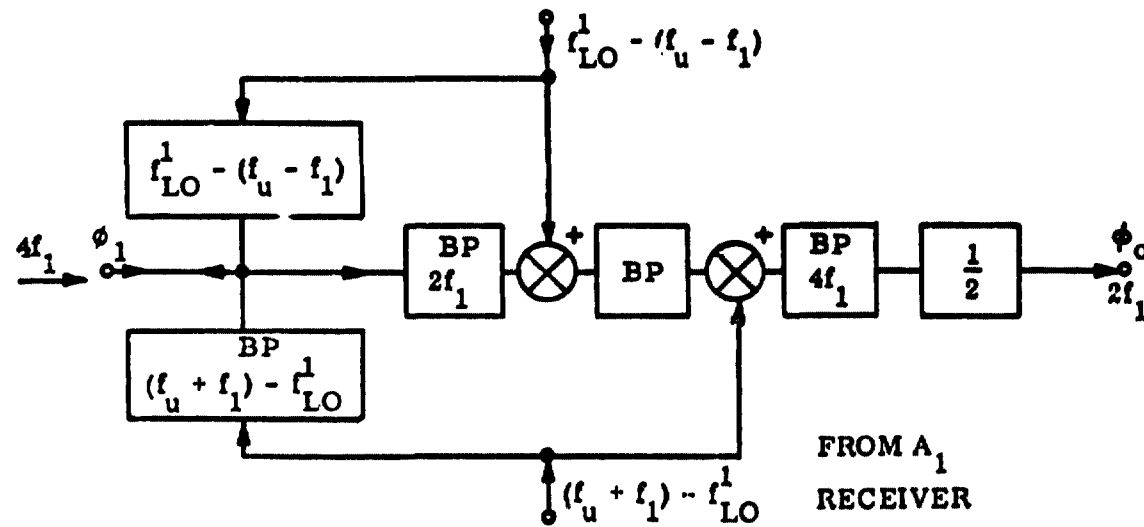
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# ALTERNATIVE PHASE REGENERATORS



SINGLE TONE PHASE REGENERATOR



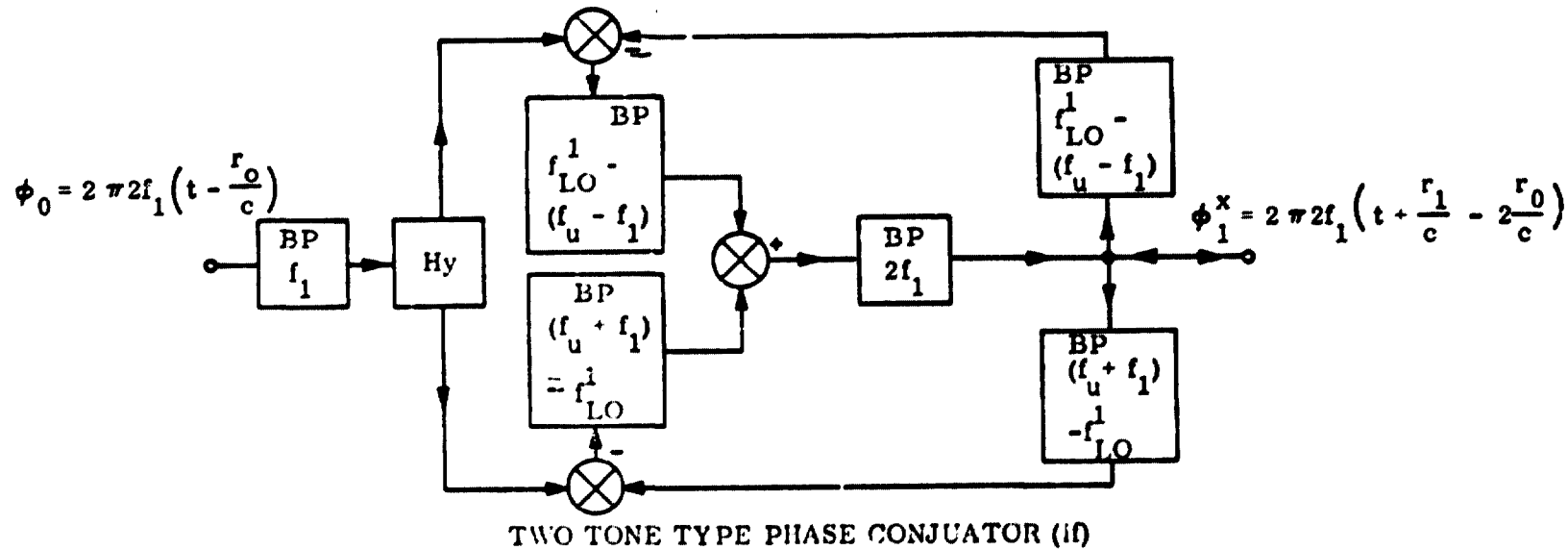
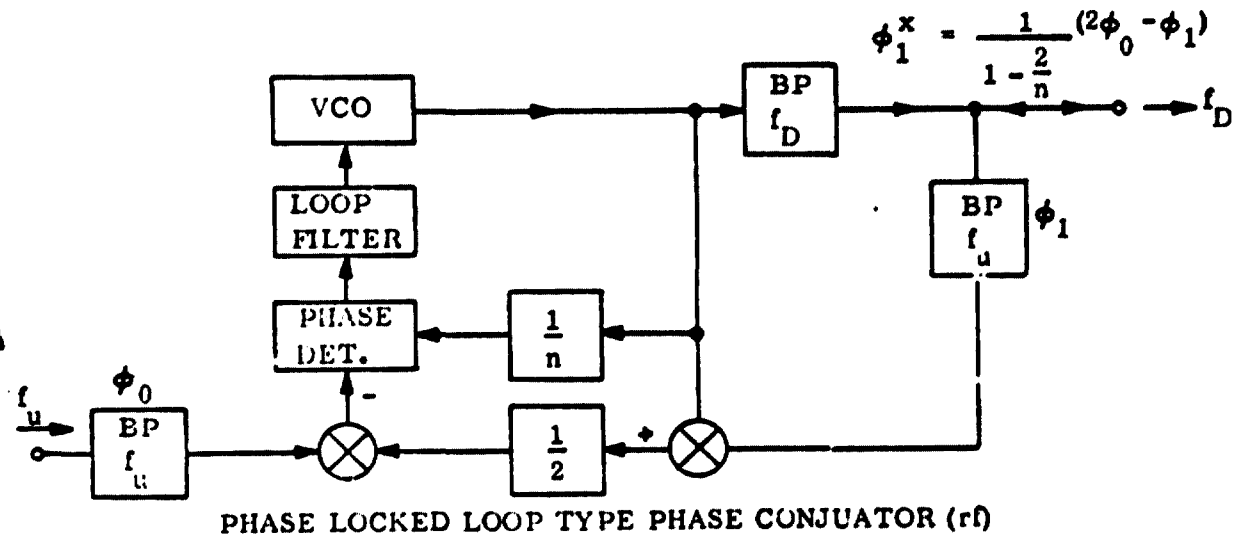
TWO TONE PHASE REGENERATOR

ALTERNATIVE PHASE CONJUGATORS

"EXACT" AND "APPROXIMATE" PHASE CONJUGATORS CAN BE USED FOR THE PRESENT PURPOSE. THE FIGURE SHOWS TWO OF THE "EXACT" TYPES OF CONJUGATORS, WHEN THE FREQUENCY TRANSLATION IS ACHIEVED WITHOUT AN ERROR IN THE CONJUGATION. CIRCUITS WHICH USE FREQUENCY DIVIDERS MUST BE SYNCHRONIZED IN ORDER TO AVOID  $\pi$  PHASE AMBIGUITY.

ALTERNATIVE PHASE CONJUGATORS

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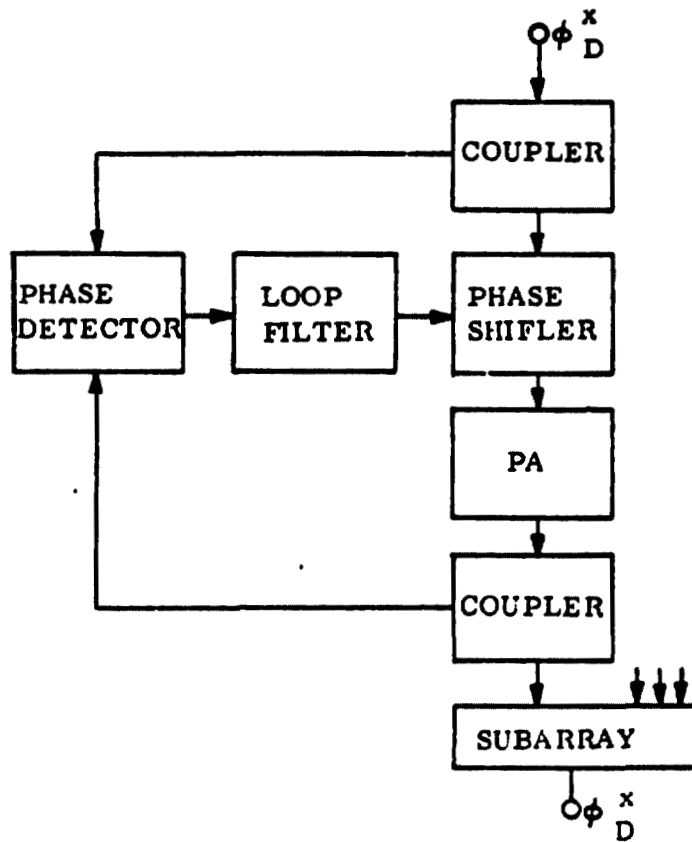


TRANSMITTER BLOCK DIAGRAM

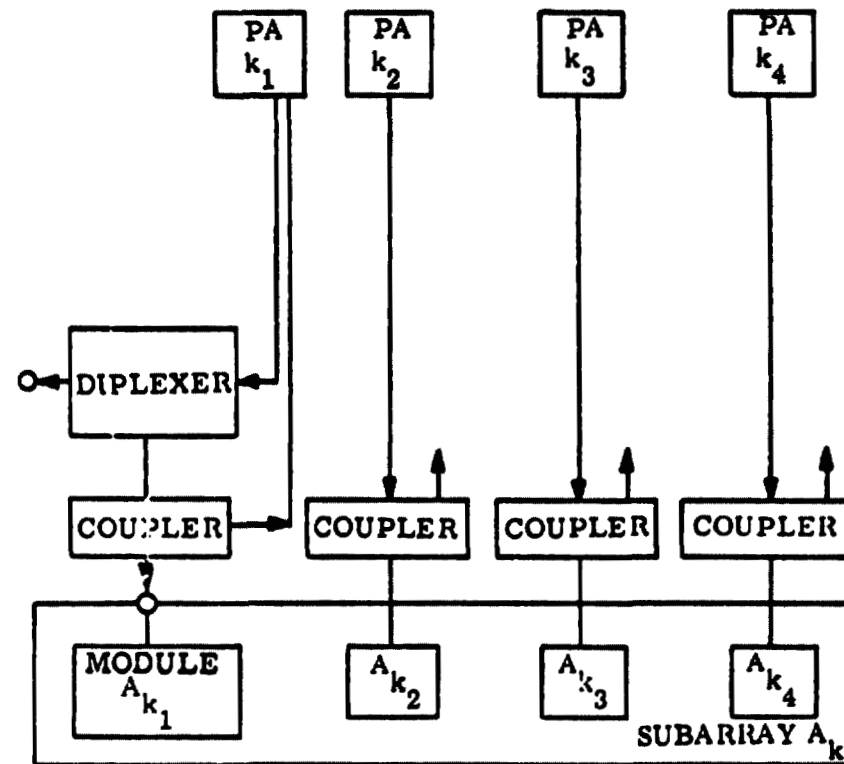
THE TRANSMITTERS OPERATING INTO A SUBARRAY ARE INDEPENDENT FROM EACH OTHER, BUT RECEIVE THEIR INPUT PHASE FROM A COMMON PHASE CONJUGATOR. ADDITIONALLY ONE TRANSMITTER MODULE SHARES A PART OF THE SUBARRAY SURFACE WITH THE RECEIVER ESTABLISHING THE PHASE FOR THE ENTIRE SUBARRAY. WHEN ELECTRONIC STEERING OF THE SUBARRAY PATTERN IS DESIRABLE FOR INCLINATION ANGLE COMPENSATION THE NORTH-SOUTH WIDTH OF A TRANSMIT ARRAY MODULE HAS TO BE RESTRICTED TO ABOUT .7 m AND THE INPUT PHASE TO THE PHASE DETECTORS OF THE TRANSMIT PHASING CIRCUIT HAS TO BE MODULATED BY A SMALL, CALCULATED ERROR SIGNAL WITH 24 HOURS PERIODICITY.



# TRANSMITTER BLOCK DIAGRAM



PHASING CIRCUIT FOR ONE HIGH  
POWER TRANSMITTER MODULE



OVERALL TRANSMITTER SYSTEM.

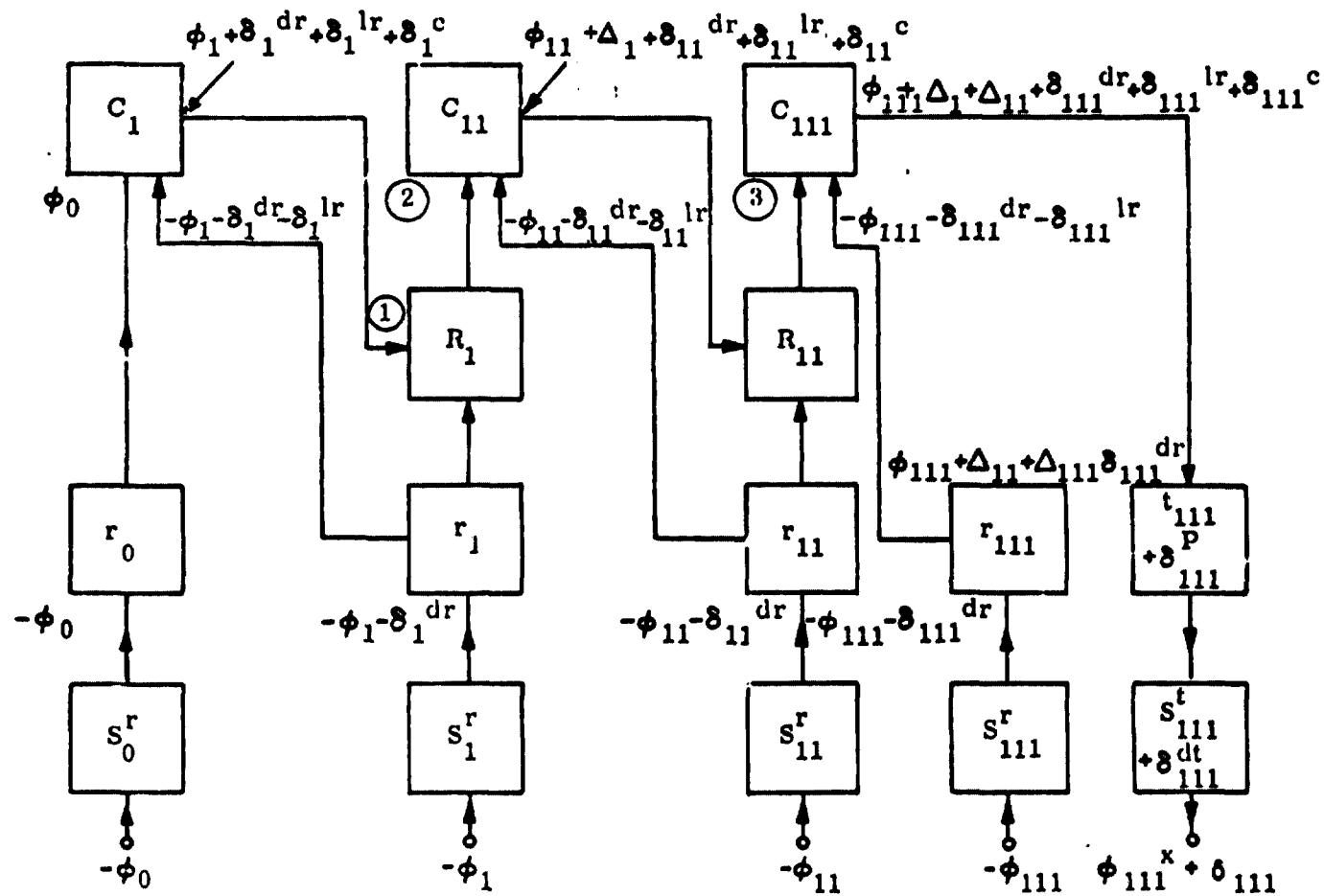
THREE LAYERS PHASE DISTRIBUTION NETWORK PHASE ERROR BUILD UP

RANDOM AMPLITUDE AND PHASE, AND SYSTEMATIC POINTING AND AMPLITUDE ERRORS ARE AFFECTING THE RESULTANT POWER TRANSFER EFFICIENCY OF ANTENNA. EFFICIENCY IS CONSIDERED BETWEEN THE OUTPUT PLANE OF THE SPACE ANTENNA AND THE INPUT PLANE OF THE RECTENNA.





### THREE LAYERS PHASE DISTRIBUTION NETWORK PHASE ERROR BUILD UP



$$\text{RSS PHASE ERROR } \delta = [3\delta_c^2 + 6\delta_1^2 + 2\delta_d^2 + \delta_p^2]^{1/2}$$

$$\textcircled{1} \phi_1 + \delta_1^{dr} + \delta_1^{lr} + \delta_1^c + \delta_1^{lt}$$

$$\textcircled{2} \phi_0 + \delta_1^{dr} + \delta_1^{lr} + \delta_1^c + \delta_1^{lt} + \delta_1^{dr} \\ = \phi_0 + \Delta_1 \quad 367$$

$$\textcircled{3} \phi_0 + \Delta_1 + \delta_{11}^{lr} + \delta_{11}^c + \delta_{11}^{lt} \\ = \phi_0 + \Delta_1 + \Delta_{11}$$

ILLUMINATION ERRORS AFFECTING ANTENNA EFFICIENCY

FOR THE SELECTED THREE LAYER PHASE DISTRIBUTION NETWORK APPROXIMATELY 95% OF ALL SUBARRAYS BELONG TO THE THIRD LAYER, THUS THEIR PHASE CONJUGATION ERRORS WILL DOMINATE THE RESULTANT PHASE CONJUGATION ERROR. THE FIGURE SHOWS THE ERROR BUILD UP FOR SUCH AN ELEMENT. THE ERROR IS A FUNCTION OF ERRORS ASSOCIATED WITH THE CONJUGATORS, CONNECTING LINE MISMATCHES, r-f DIPLEXER DIFFERENTIALS AND ERRORS ASSOCIATED WITH THE TRANSMITTER PHASING CIRCUIT OF ALL THE ELEMENTS WHICH ARE ASSOCIATED WITH THE PARTICULAR TRANSMIT SUBARRAY.


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## ILLUMINATION ERRORS AFFECTING ANTENNA EFFICIENCY

| <u>RANDOM</u>                                           |                       | <u>SYSTEMATIC</u>               |                          |
|---------------------------------------------------------|-----------------------|---------------------------------|--------------------------|
| <u>PHASE</u>                                            | <u>AMPLITUDE</u>      | <u>POINTING</u>                 | <u>AMPLITUDE</u>         |
| PHASE JITTER ( $f_u, f_l$ )                             | <u>TRANSMIT POWER</u> | DOPPLER FREQUENCY               | ILLUMINATION             |
| TRANSMITTER NOISE                                       | SUBARRAY ROTATION     | SHIFT                           | QUANTIZATION             |
| CONJUGATOR ( $\delta_c$ )                               |                       | ABERRATION                      | POLARIZATION<br>ROTATION |
| <u>LINE MATCH DIFFERENTIALS (<math>\delta_d</math>)</u> |                       | <u>IONOSPHERIC DIFFERENTIAL</u> |                          |
| DIPLEXER MATCH DIFFERENTIALS ( $\delta_d$ )             |                       | <u>ATMOSPHERIC DIFFERENTIAL</u> |                          |
| TRANSMITTER PHASING ( $\delta_p$ )                      |                       |                                 |                          |
| DIFFERENTIAL DOPPLER                                    |                       |                                 |                          |

RANDOM ERRORS FOR 3 LAYERS PHASE DISTRIBUTION

THE RANDOM PHASE ERRORS ARE DOMINATED BY THE 1-f DIPLEXER IMPERFECTIONS. THE AMPLITUDE ERRORS ARE DOMINATED BY ARRAY TILT.



## RANDOM ERRORS FOR 3 LAYERS PHASE DISTRIBUTION

| <u>SOURCE</u>                            | <u>PHASE ERRORS (deg)</u>      | <u>DEG.</u>        |
|------------------------------------------|--------------------------------|--------------------|
| PHASE JITTER                             |                                | 1.13               |
| TRANSMITTER NOISE ( $c/n = 30$ db)       |                                | .36                |
| CONJUGATORS ( $\delta_c = .6^\circ$ )    |                                | 1.04               |
| LINEs ( $\delta_l = 2.54^\circ$ )        |                                | 6.22               |
| DIPLEXERS ( $\delta_d = 1.81^\circ$ )    |                                | 2.56               |
| TRANSMITTERS ( $\delta_p = 1.6^\circ$ )  |                                | 1.60               |
| DIFFERENTIAL DOPPLER ( $V_d = 6.25$ m/s) |                                | .18                |
| <hr/>                                    |                                |                    |
|                                          | PEAK:                          | 13.09    RSS: 7.09 |
|                                          | PHASE ERROR CAUSED LOSS: 1.53% |                    |

| <u>SOURCE</u>                                                 | <u>PEAK</u>                      | <u>RMS</u> |
|---------------------------------------------------------------|----------------------------------|------------|
| TRANSMIT POWER FLUCTUATION (1 db, rms)                        | 10.64                            | 2.38       |
| ARRAY ROTATION ( $L_s \leq 10m, \Delta\theta_s = .15^\circ$ ) | 13.50                            | 1.41       |
| <hr/>                                                         |                                  |            |
|                                                               | PEAK: 24.14                      | RSS: 2.51  |
|                                                               | AMPLITUDE ERROR CAUSED LOSS:     |            |
|                                                               | For $\Delta\theta_s = .15^\circ$ | 2.51%      |
|                                                               | $\Delta\theta_s = .05^\circ$     | 1.34%      |

SYSTEMATIC ERRORS FOR 3 LAYERS PHASE DISTRIBUTION

THE SYSTEMATIC ERRORS ARE DOMINATED BY PROPAGATION ERRORS. ALTHOUGH THESE MAY SHOW UP IN A SMALL PERCENTAGE OF TIME ONLY THEIR VALUE CAN BE SIGNIFICANT IF ONLY ONE PILOT ANTENNA IS USED.



# SYSTEMATIC ERRORS FOR 3 LAYERS PHASE DISTRIBUTION

## POINTING ERRORS (deg)

| SOURCE                                                                      | 1 PILOT STATION                                         | 3 PILOT STATION                                  |
|-----------------------------------------------------------------------------|---------------------------------------------------------|--------------------------------------------------|
| DOPPLER ( $i = 2.2^\circ$ , $r_m^i = 13.6$ m/s, $2 f_{Dop}^i = 112$ Hz)     | $1.43 \times 10^{-6}$                                   | $7.15 \times 10^{-8}$                            |
| ABERRATION ( $\dot{Z}_m^i = 100$ m/s)                                       | $19.3 \times 10^{-6}$                                   | $9.65 \times 10^{-8}$                            |
| IONOSPHERIC DIFFERENTIAL ( $.1^\circ$ way refraction)                       | $2.35 \times 10^{-3}$                                   | $1.17 \times 10^{-4}$                            |
| ATMOSPHERIC DIFFERENTIAL ( $.3^\circ$ 1 way refraction,<br>2% irregularity) | $6.00 \times 10^{-3}$                                   | $3.00 \times 10^{-4}$                            |
| Pointing error (deg)                                                        | PEAK $8.35 \times 10^{-3}$<br>RSS $6.44 \times 10^{-3}$ | $4.175 \times 10^{-4}$<br>$3.221 \times 10^{-4}$ |
| Pointing loss (%)                                                           | PEAK 1.19<br>RSS .92                                    | .60<br>.46                                       |

## AMPLITUDE ERRORS (%)

### QUANTIZATION

|                               |      |
|-------------------------------|------|
| 16 LEVEL DISTRIBUTION         | .078 |
| 8 LEVEL DISTRIBUTION          | .312 |
| FARADAY ROTATION (WORST YEAR) | .48  |

SUMMARY OF LOSSES

NET LOSS IN THE ANALYZED SECTION OF THE OVERALL SYSTEM CAN BE KEPT A 3.65% RMS LEVEL. FARADAY ROTATION LOSS, ALTHOUGH IT IS SMALL, CAN BE FURTHER REDUCED BY ABOUT A FACTOR OF TWO ON THE AVERAGE, IF THE POLARIZATION ANGLE OF THE SPACE ANTENNNA IS CORRECTED ONCE A YEAR.



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SUMMARY OF LOSSES

| <u>SOURCE</u>                                 | <u>LOSS (%)</u>          |
|-----------------------------------------------|--------------------------|
| RANDOM PHASE                                  | 1.53                     |
| RANDOM AMPLITUDE                              | 1.34                     |
| SYSTEMATIC POINTING (3 PILOT STATION)         | .46                      |
| SYSTEMATIC AMPLITUDE (8 LEVELS)               | .32                      |
| <hr/>                                         |                          |
| RESULTANT LOSS ASSOCIATED TO SPACECRAFT ARRAY | 3.65, RMS                |
| FARADAY ROTATION (HOUSTON, WORST YEAR)        | .48%, "AVERAGE"<br>PEAK. |

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MULTI TONE PHASE COMPUTING SPS PHASE CONTROL SYSTEM

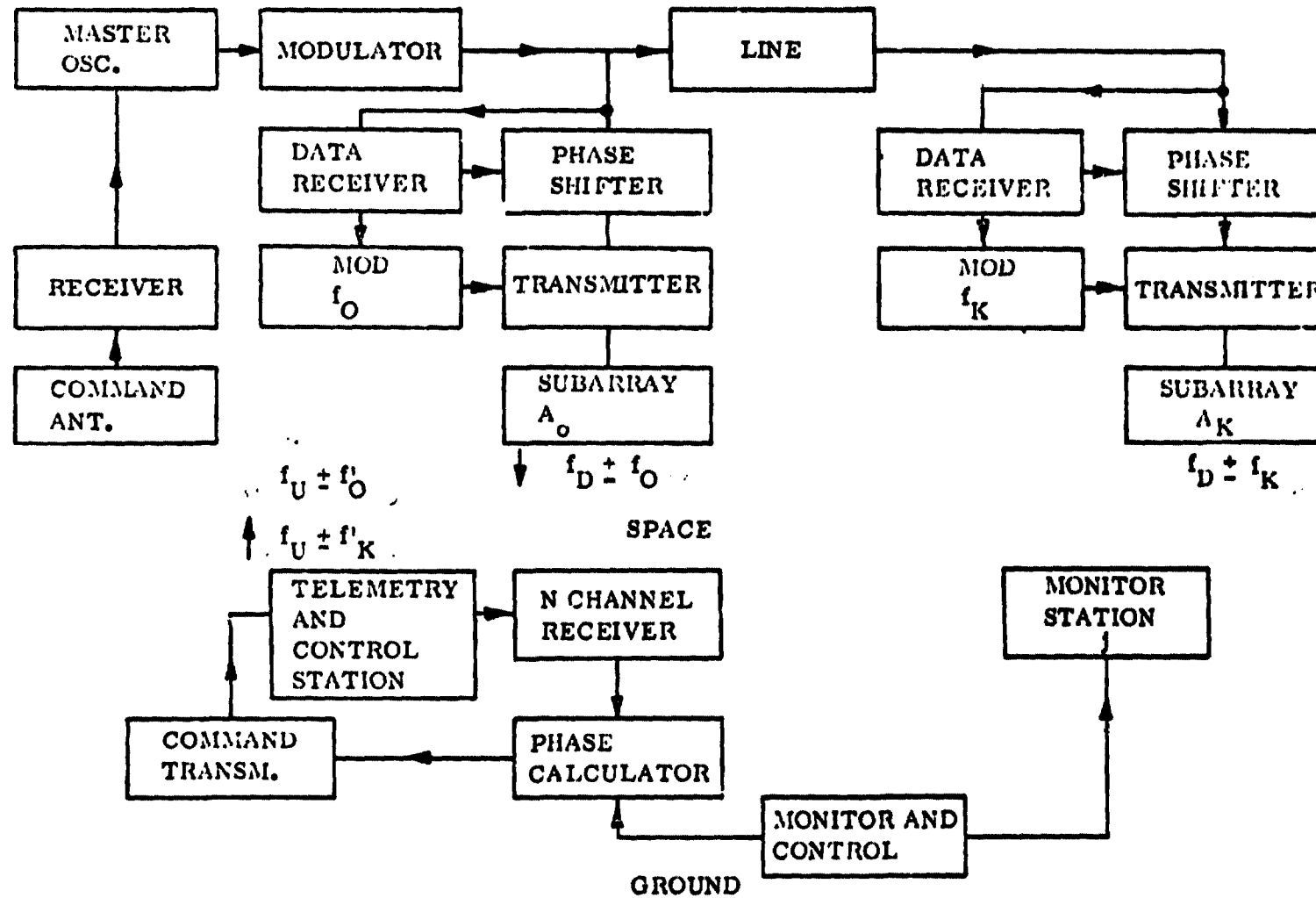
THE MULTITONE PHASE COMPUTING SYSTEM IS BASED ON THE MEASUREMENT OF THE PHASE OF EACH SUBARRAY ON THE GROUND BY THE USE OF A SIGNATURE TONE ASSOCIATED WITH THE SUBARRAY. THE REQUIRED PHASE CORRECTION FOR THE SUBARRAY TRANSMITTER IS RETURNED BY THE USE OF AN UPLINK CONTROL CHANNEL. BOTH FREQUENCY AND TIME DIVISION IS USED TO REDUCE COMPLEXITY AND REQUIRED FREQUENCY BAND FOR THE TONES. APPROXIMATELY 25 MHz BANDWIDTH IS NEEDED FOR A 1 SEC PHASE UPDATING PERIOD.



# MULTI TONE PHASE COMPUTING SPS PHASE CONTROL SYSTEM

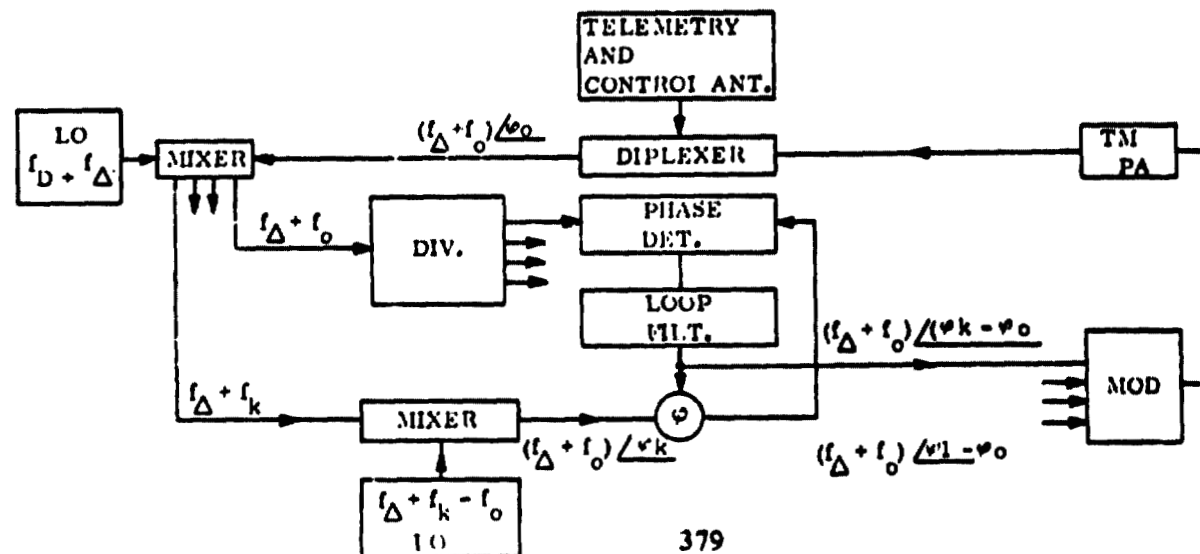


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SYSTEM BLOCK DIAGRAM OF SPS PHASE CONTROL CIRCUIT IN PHASE  
COMPUTING MODE OF OPERATION (N TONE DOWNLINK)

THE IMPLEMENTATION OF THE N TONE PHASE COMPUTING SYSTEM REQUIRES A SINGLE TELEMETRY-CONTROL ANTENNA ON THE GROUND CLOSE TO THE MIDDLE OF THE RECTENNA AND ON N TONE RECEIVER SYSTEM. THE POWER BEAM AT THE SPACECRAFT TRANSMITTER IS MODULATED BY A LOW LEVEL TONE. THESE TONES ARE DETECTED ON THE GROUND AND THEIR PHASE IS COMPARED AGAINST THE PHASE OF AN ARBITRARILY SELECTABLE REFERENCE TONE. A TYPICAL SYSTEM MAY USE 100 TONES, 100 TIME DIVISION CHANNELS AND 1 SEC FOR THE COMPLETE PHASE UPDATING OF THE TRANSMITTERS.



SUMMARY

A SUITABLE PHASE CONTROL SYSTEM CAN BE ESTABLISHED EITHER ON THE BASIS OF GROUND OR SPACE ORIGINATED PILOT SIGNALS. THE RESULTANT POWER TRANSFER EFFICIENCY IS A FUNCTION OF A LARGE NUMBER OF PARAMETERS, BUT A REALISTIC SET OF SYSTEM PARAMETERS RESULTS IN A TYPICAL 3.65% EFFICIENCY ASSOCIATED WITH THE PHASE CONTROL SYSTEM.



SUMMARY

- SEVERAL FEASIBLE SYSTEMS ARE ESTABLISHED
- RETRODIRECTIVE SYSTEM-COMPLEXITY IN SPACE
- PHASE COMPUTING SYSTEM-COMPLEXITY ON THE GROUND
- RETRODIRECTIVE SYSTEM HAS FAST RESPONSE, ACCEPTABLE ACCURACY BUT MAY BE MORE COSTLY AND LESS RELIABLE.
- MORE WORK IS NEEDED ON PHASE COMPUTING SYSTEM BEFORE FAIR COMPARISON CAN BE MADE.
- TRANSMISSION LOSS IN RETRODIRECTIVE SYSTEM CAN BE LESS THAN 3.65% WITHOUT FARADAY ROTATION LOSS.
- MAJOR LOSS FACTORS: TRANSMIT POWER FLUCTUATION, MISMATCHES, PROPAGATION ERRORS.
- FARADAY LOSS COULD REACH .48% DAILY PEAK IN WORST YEAR FOR STANDARD IONOSPHERE.
- EXPERIMENTAL WORK IS RECOMMENDED TO IMPROVE PHASE ERROR PREDICTION OF SUBSYSTEMS.
- UNIT COUNT IS OVER  $10^6$  IN RETRO SYSTEM. (TYPICAL UNITS: INTEGRATED CIRCUITS, PA TUBE, SUBARRAY PANEL, ETC.)
- CAPITAL INVESTMENT PER WATT CAPACITY VARIES SLOWLY FOR 6-8 Gw OUTPUT POWER RANGE.

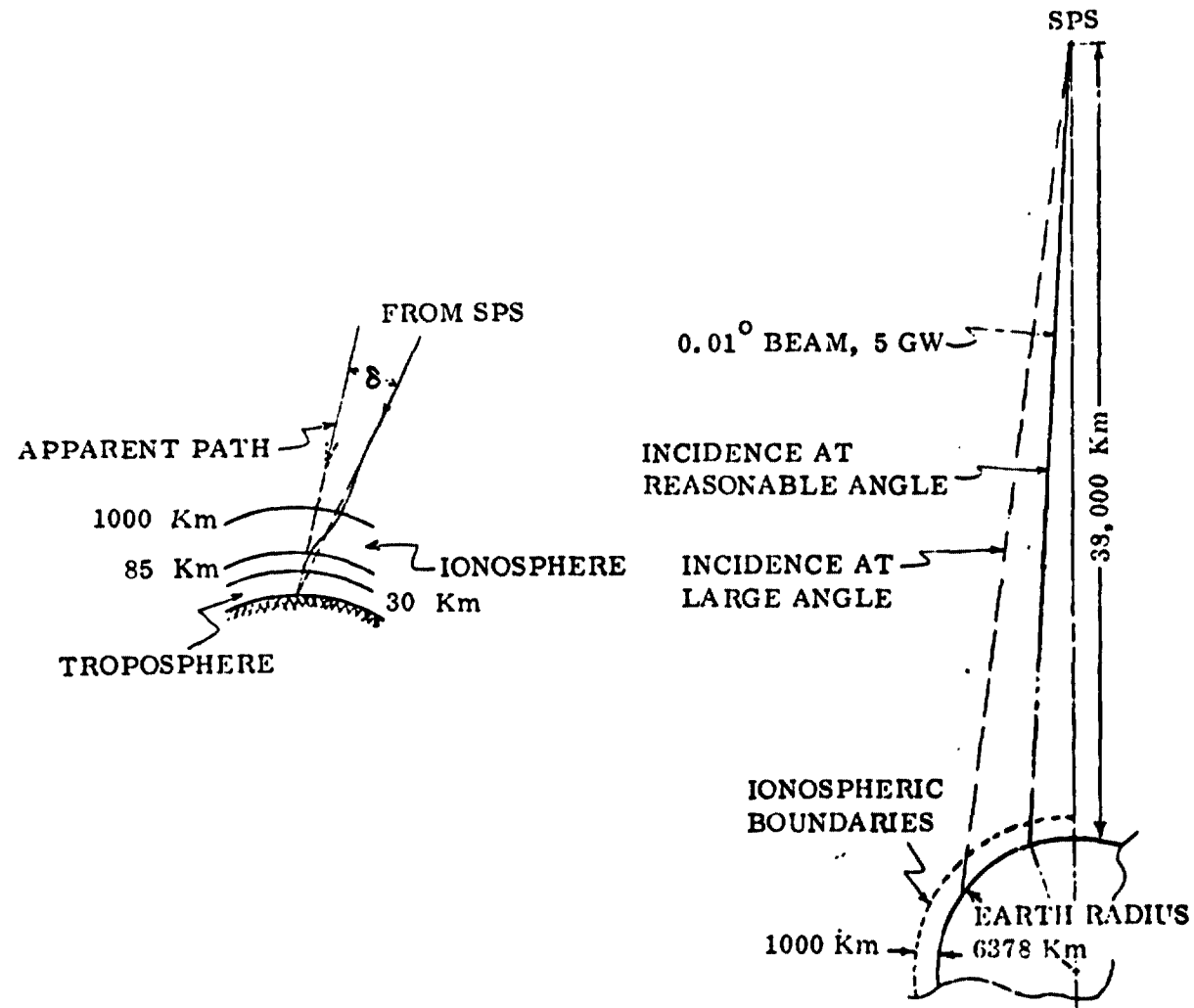
GEOMETRY OF SPS PROPAGATION PATH

THE PENETRATION PATH THROUGH THE ATMOSPHERE AND THROUGH THE IONOSPHERE IS SMALL RELATIVE TO THE TOTAL PATH LENGTH. THESE YIELD MAXIMUM BENDING EFFECT, DUE TO THE ERROR ANGLE  $\delta$ , OF APPROXIMATELY  $0.02^\circ$  (WHICH CORRESPONDS TO BEAM DEVIATION OF LESS THAN 10 METERS FROM THE CENTER OF THE RECTENNA).





# GEOMETRY OF SPS PROPAGATION PATH



SUMMARY OF ATMOSPHERIC EFFECTS ON SPS RADIATION PATTERN

- ATMOSPHERIC REFRACTION EFFECTS ARE NEGLIGIBLE.
- ATMOSPHERIC ABSORPTION IS APPROXIMATELY TWO PERCENT.
- 2.45 GHz IS PREFERABLE THAN 5.8 GHz BECAUSE OF  
DRASTIC RAIN ATTENUATION AT 5.8 GHz.



**SUMMARY OF ATMOSPHERIC EFFECTS ON  
SPS RADIATION PATTERN**



| Cause                                       | Effect                                                     |          |          |
|---------------------------------------------|------------------------------------------------------------|----------|----------|
| Absorption                                  | < 0.02%                                                    |          |          |
| Accumulated<br>Phase Front<br>Perturbations | Elevation<br>90°<br>(Zenith)                               | 2.45 GHz | 5.8 GHz  |
|                                             |                                                            | 106°     | 45°      |
|                                             | Elevation<br>20°                                           | 162°     | 69°      |
| Scintillation<br>Attenuation<br>Effects     | Elevation<br>90°<br>(Zenith)                               | 0.785 dB | 0.35 dB  |
|                                             | Elevation<br>20°                                           | 1.172 dB | 0.528 dB |
| Scattering                                  | -27 dB Sidelobe level relative to the<br>peak of radiation |          |          |

SUMMARY OF IONOSPHERIC EFFECTS ON SPS RADIATION PATTERN

IONOSPHERIC ABSORPTION IS NEGLIGIBLY SMALL.

MAXIMUM PHASE FRONT ERROR DUE TO PERTURBED IONOSPHERE  
IS  $162^{\circ}$  AT ELEVATION OF  $20^{\circ}$  AT A RATE OF CHANGE OF  
APPROXIMATELY  $18^{\circ}$ /SECOND.

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# SUMMARY OF IONOSPHERIC EFFECTS ON SPS RADIATION PATTERN



| Cause                                    | Effect                                         |                                                        |          |         |
|------------------------------------------|------------------------------------------------|--------------------------------------------------------|----------|---------|
| Refraction,<br>Unperturbed<br>Atmosphere | Elevation<br>20°                               | Elevation error angle of 0.03° ± 0.006° on cloudy days |          |         |
|                                          |                                                | Elevation error angle of 0.02° ± 0.002° on clear days  |          |         |
|                                          | Elevation<br>90°<br>(Zenith)                   | Elevation error angle of ± 0.006° on cloudy days       |          |         |
|                                          |                                                | Elevation error angle of + 0.002° on clear days        |          |         |
| Refraction,<br>Perturbed<br>Atmosphere   | ± 0.0036° peak angle of arrival fluctuations   |                                                        |          |         |
|                                          | ± 10° peak phase error in incoming phase front |                                                        |          |         |
| Absorption,<br>Without Rain              | Elevation<br>90°                               | Humidity                                               | 2.45 GHz | 5.8 GHz |
|                                          |                                                | 0%                                                     | 1.2%     | 1.3%    |
|                                          | 100%                                           | 1.2%                                                   | 1.6%     |         |
|                                          | Elevation<br>20°                               | 0%                                                     | 1.4%     | 1.5%    |
|                                          |                                                | 100%                                                   | 1.4%     | 2%      |
|                                          | 30 mm/hr.<br>Rain<br>Absorption                | Elevation<br>90°                                       | 100%     | .42%    |
|                                          | Elevation<br>20°                               | 100%                                                   | 1.25%    | 33.6%   |

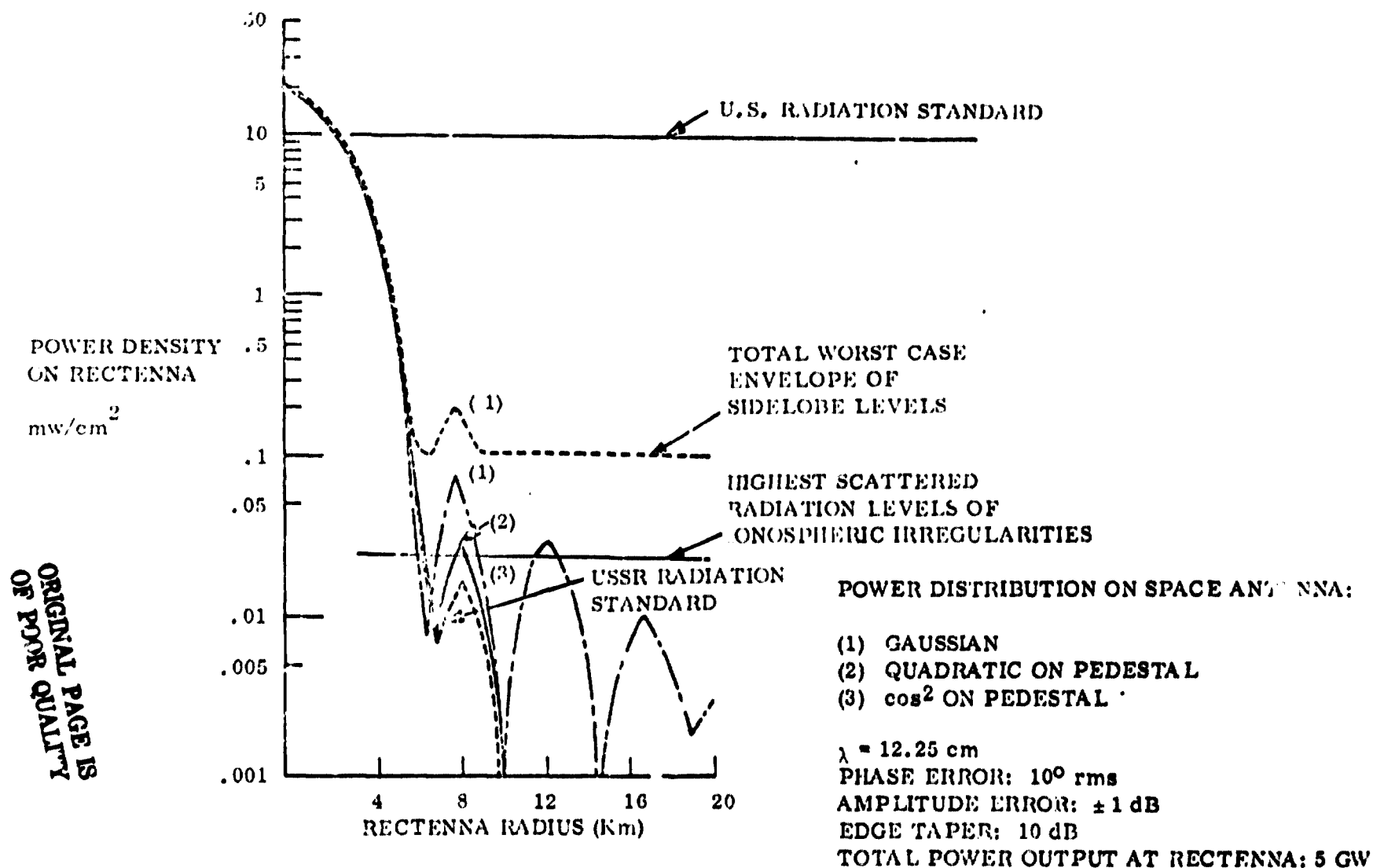
EFFECT OF PROPAGATION MEDIA ON SIDELobe LEVEL  
(VARIOUS POWER DISTRIBUTIONS ON SPACE ANTENNA)

MAXIMUM SCATTERED RADIATION LEVELS NEAR THE MAIN BEAM OCCUR WHEN THE PHASE ERRORS OF THE WAVEFRONT REACH THE VALUE OF ONE RADIAN. THE ENVELOPE OF THESE PEAK LEVELS IS SHOWN AT APPROXIMATELY 27 DB BELOW THE PEAK LEVEL OF RADIATION. WHEN THESE LEVELS INTERACT WITH THE WORST SIDELobe LEVELS OF CURVE (1) (GAUSSIAN TRANSMIT BEAM WITH 10 DB TAPER) THE WORST CASE OF INTERACTION ENVELOPE (1) APPEARS AS SHOWN BY DOTTED LINES.



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# EFFECT ON PROPAGATION MEDIA ON SIDELobe LEVEL



POLARIZATION ROTATION VS. FREQUENCY DUE TO IONOSPHERE  
(TYPICAL VALUE FOR CONTINENTAL US IN WORST YEAR)

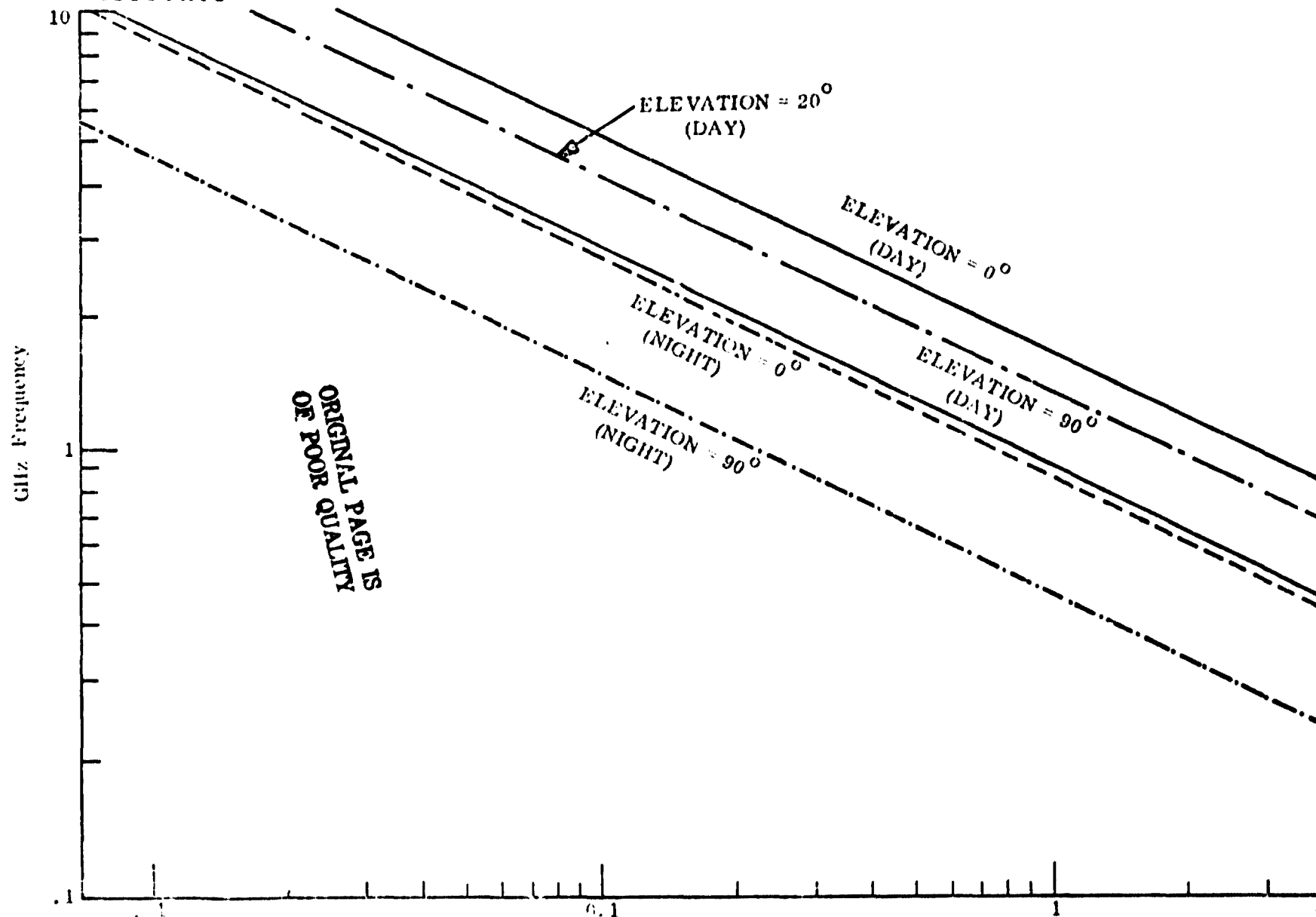
LOSS OF POWER DUE TO THE FARADAY ROTATION OF  $17.2^{\circ}$  AT  
2.45 GHz (WITH RECTENNA POLARIZATION ADJUSTED FOR MID  
RANGE OF ROTATION) IS 1.1 PERCENT DURING THE DAY TIME.  
(THE IONOSPHERIC MAGNETIC FIELD IS ASSUMED TO BE 0.62  
GAUSS FOR THESE COMPUTATIONS).





GENERAL  
ELECTRIC

D180-24071-3  
POLARIZATION ROTATION VS FREQUENCY  
DUE TO IONOSPHERE



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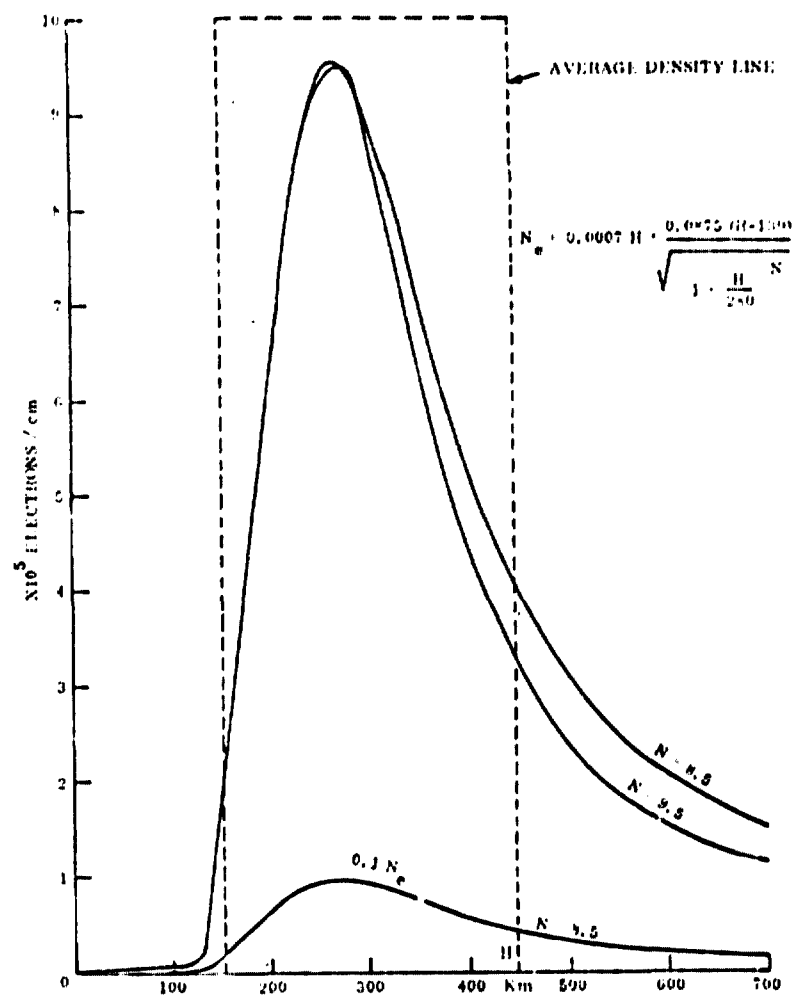
VARIATION OF ELECTRON DENSITY IN THE IONOSPHERE WITH HEIGHT

GENERAL ELECTRIC MATHEMATICAL MODEL OF THE IONOSPHERIC ELECTRON DENSITY AS A FUNCTION OF HEIGHT.



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# VARIATION OF ELECTRON DENSITY IN THE IONOSPHERE WITH HEIGHT



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ATTENUATION VS. FREQUENCY IN THE IONOSPHERE

IONOSPHERIC ATTENUATION IS INVERSELY PROPORTION TO THE SQUARE OF THE FREQUENCY.

THIS AMOUNTS TO LESS THAN 0.02% AT ALL FREQUENCIES OF INTEREST AND AT WORST CONDITIONS OF PROPAGATION.

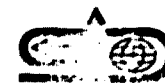
SCATTERED RADIATION DUE TO THE CROSS SECTION OF THE ELECTRONS IS QUITE SMALL RELATED TO THE SIDELobe LEVELS OF THE RECTENNA. THESE LEVELS MIGHT, HOWEVER, BE HARMFUL FOR INTERFERENCE WITH OTHER COMMUNICATION SYSTEMS.



GENERAL  
ELECTRIC

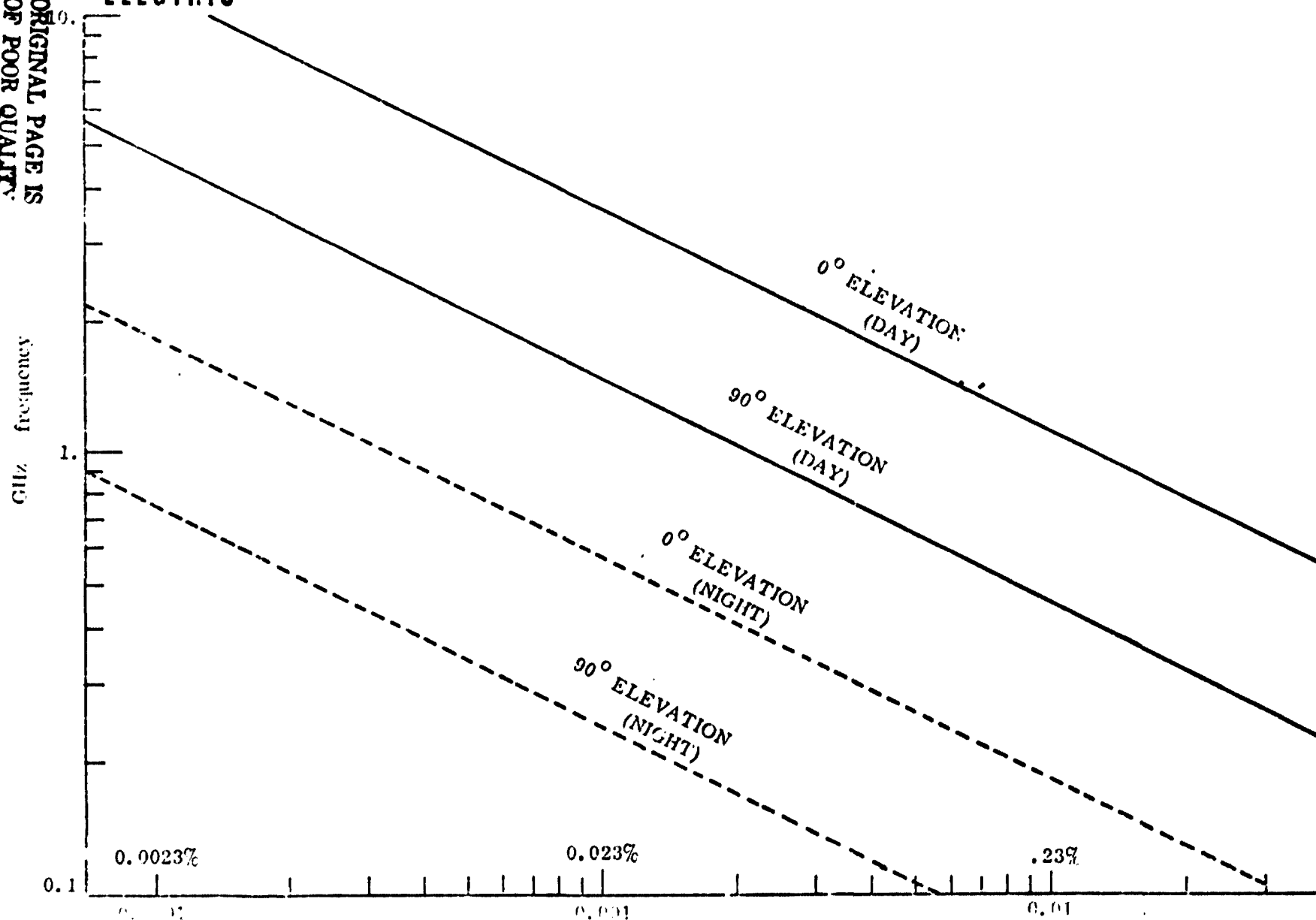
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# ATTENUATION VS FREQUENCY IN THE IONOSPHERE



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RECTENNA/GROUND POWER COLLECTION & TRANSMISSION

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RECTENNA TECHNOLOGY BASIC ASSUMPTIONS  
(5 GW SYSTEM)

THE IMPORTANT DESIGN CONSIDERATIONS OF THE RECTENNA ARE DETERMINED FROM:

1. THE CAPABILITY OF THE RF/DC CONVERTERS TO WORK IN HARMONY AT THE PROJECTED LEVELS OF EFFICIENCY AT DIFFERENT LEVELS OF RF DENSITIES OF THE RECTENNA STRUCTURE.
2. THE STRUCTURAL CAPABILITY TO MINIMIZE THE DIFFRACTION SHADOWING FACTOR K, AS FUNCTION OF THE LATITUDE LOCATION, IN ORDER TO MINIMIZE THE FIELD LEVEL VARIATIONS ACROSS THE COLLECTING PANELS.



RECTENNA TECHNOLOGY BASIC ASSUMPTIONS



- DETECTION EFFICIENCY (RF/DC) 85 - 92 %
- RANGE OF POWER DENSITIES 1 - 40 MW/CM<sup>2</sup>
- QUALITY PERFECTED DIODES  $\phi$  1 / DIODE
- PRIMARY UNIT CAPACITY 1 MW
- PRIMARY UNIT D.C. VOLTAGE (5000 UNITS)  
 $\pm$  2 KV
- CONVERTER UNIT CAPACITY (2 x 20) 40 MW  
(125 UNITS)
- PANEL WIDTH (W) IS A FUNCTION OF LATITUDE  
LOCATION AS DETERMINED BY DIFFRACTION SHADOWING  
FACTOR (K) AND PANEL SEPARATION (H)  $W^2 > \frac{H \lambda}{K^2}$
- SUPPRESSION OF HARMONIC RADIATION TO ACCEPTABLE  
LEVELS



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CHARACTERISTICS OF POWER FLOW

- RADIAL FLOW OF POWER AMONG THE PRIMARY UNITS MINIMIZES DISSIPATION LOSSES, IN CONTRAST TO THE CIRCULATORY FLOW OF POWER. THE FORMER NEEDS DESIGN CARE FOR THE EQUAL PRIMARY UNITS AT DIFFERENT FIELD LEVELS.
- SLIGHTLY DIFFERENT GROUPING OF ELEMENTS OR SLIGHTLY DIFFERENT ELEMENTS ARE NEEDED ANY WAY FOR PROPER POWER FLOW IN THE RECTENNA.



CHARACTERISTICS OF POWER FLOW

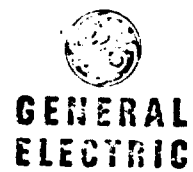


- EQUAL POWER LEVELS ARE CONCENTRIC ELLIPTIC RINGS
- THE AREA OF COLLECTING RINGS FOR A SPECIFIC AMOUNT OF POWER INCREASES EXPONENTIALLY TO THE EDGE OF THE RECTENNA
- POWER FLOW IS RADIAL FROM OR TO THE CENTER OF THE RECTENNA
- IONOSPHERIC DISTURBANCES INDUCE SLOW OSCILLATORY FORM TO THE POWER FLOW. THIS EFFECT IS RECTIFIED BY RADIAL FLOW OF POWER
- CROWBARS AND D.C. BREAKERS ARE USED FOR EACH PRIMARY UNIT OF 1 MW
- FAULTY SECTIONS MAY BE ELECTRICALLY BYPASSED BY LARGE CAPACITY DIODES PRIOR TO D.C. BREAKER ISOLATION

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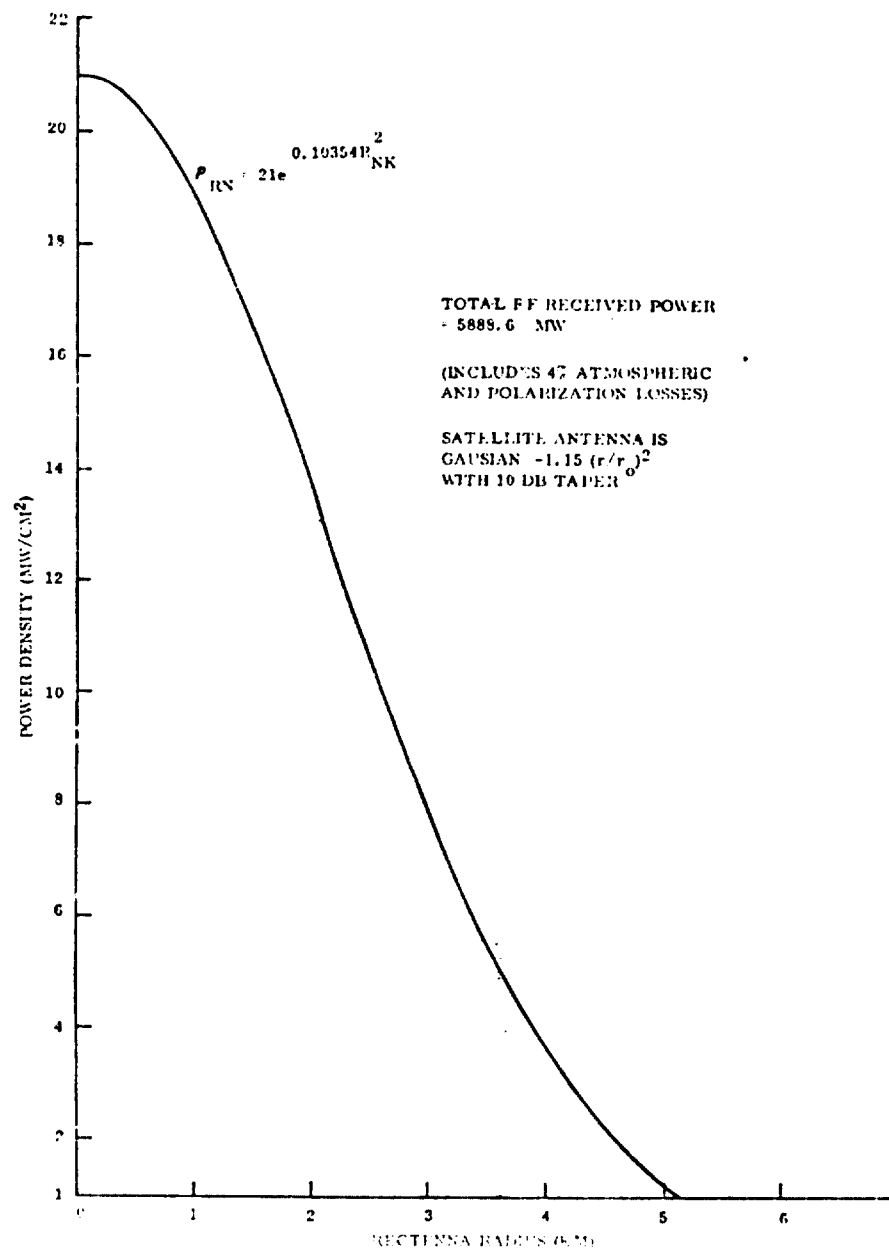
RF POWER DISTRIBUTION OF RECTENNA

THIS CHART SHOWS THE r-f POWER DISTRIBUTION OF THE RECTENNA FOR A 5 GW DC OUTPUT POWER.  
A GAUSSIAN POWER DISTRIBUTION WITH 10 DB EDGE TAPER IS ASSUMED ON SPACE ANTENNA. ALSO  
4% LOSS DUE TO MEDIA IS ASSUMED.



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RF POWER DISTRIBUTION OF RECTENNA



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PERCENTAGE OF ACCUMULATED POWER ON 10 KM DIAMETER RECTENNA

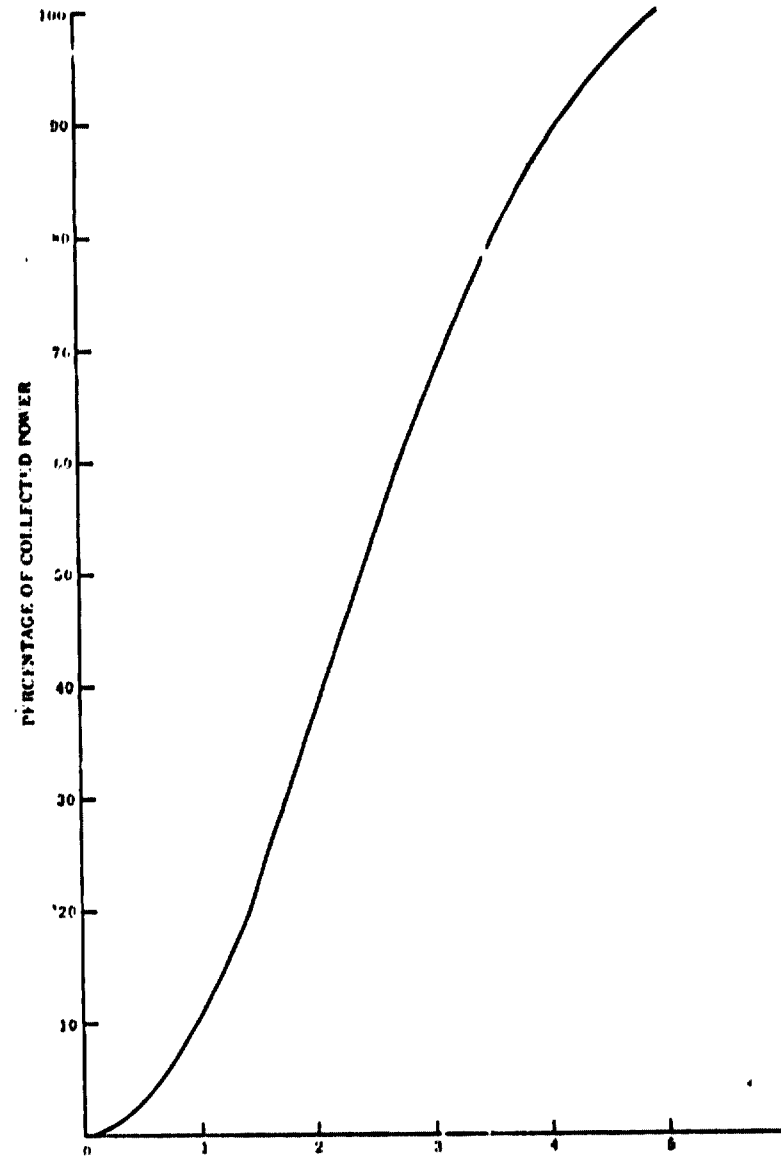
THIS FIGURE SHOWS THE TOTAL COLLECTED POWER FOR A GIVEN DOWNLINK BEAM ( 1 KM DIAMETER SPACE ANTENNA, GAUSSIAN DISTRIBUTION, 10 DB TAPER) AS A FUNCTION OF THE RADIUS OF RECTENNA.

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# PERCENTAGE OF ACCUMULATED POWER ON 10KM DIAMETER RECTENNA



RECTENNA DIAMETER (KM)

RECTENNA COLLECTING RING AREA AND NUMBER OF RADIATING  
ELEMENTS AS A FUNCTION OF RADIUS

THE POWER OF THE RECTENNA IS COLLECTED IN SEGMENTS CORRESPONDING TO CONCENTRIC RINGS. THE AREA OF THESE RINGS INCREASES WITH THE RING RADIUS IN ORDER TO KEEP THE POWER OUTPUT FROM THE SEGMENTS CONSTANT.

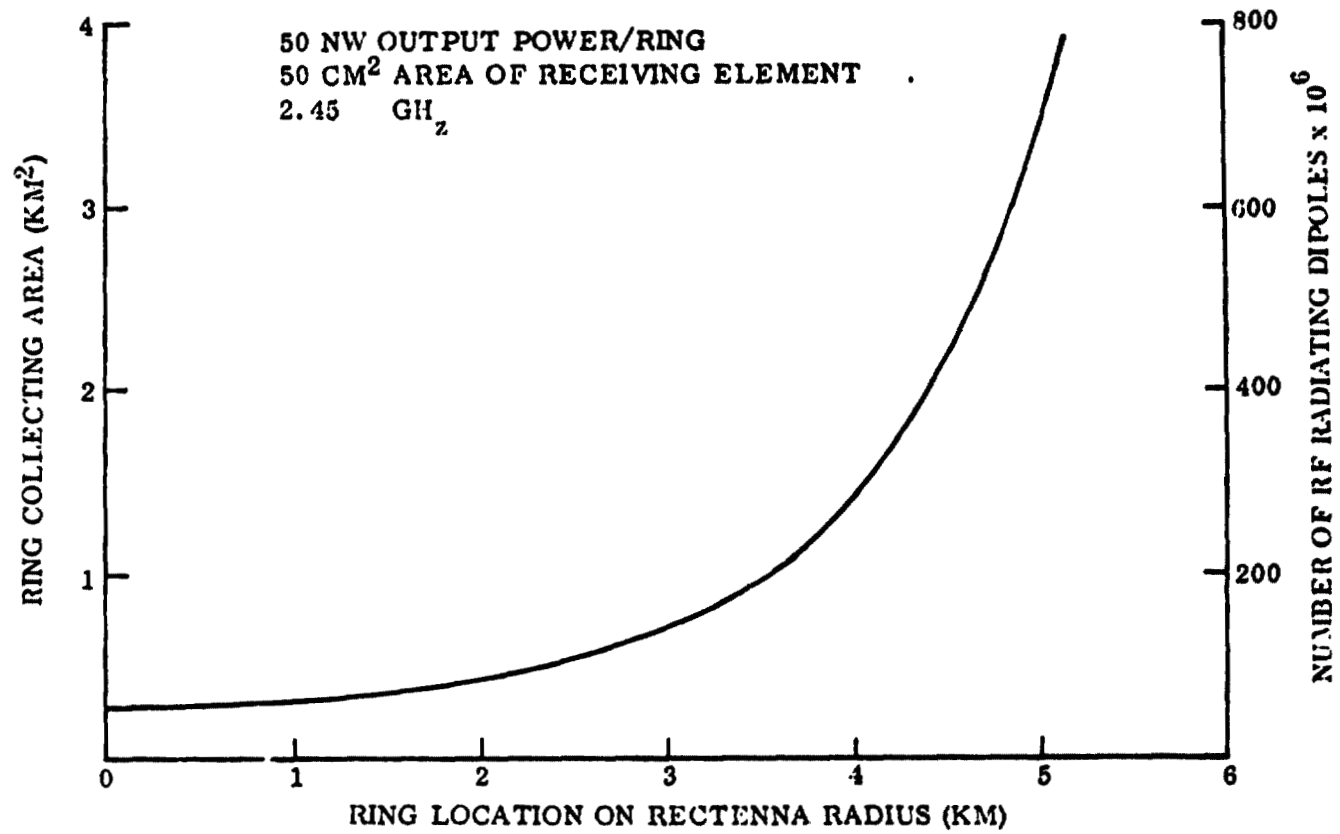
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RECTENNA COLLECTING RING AREA AND NUMBER OF  
RADIATING ELEMENTS AS FUNCTION OF RADIUS



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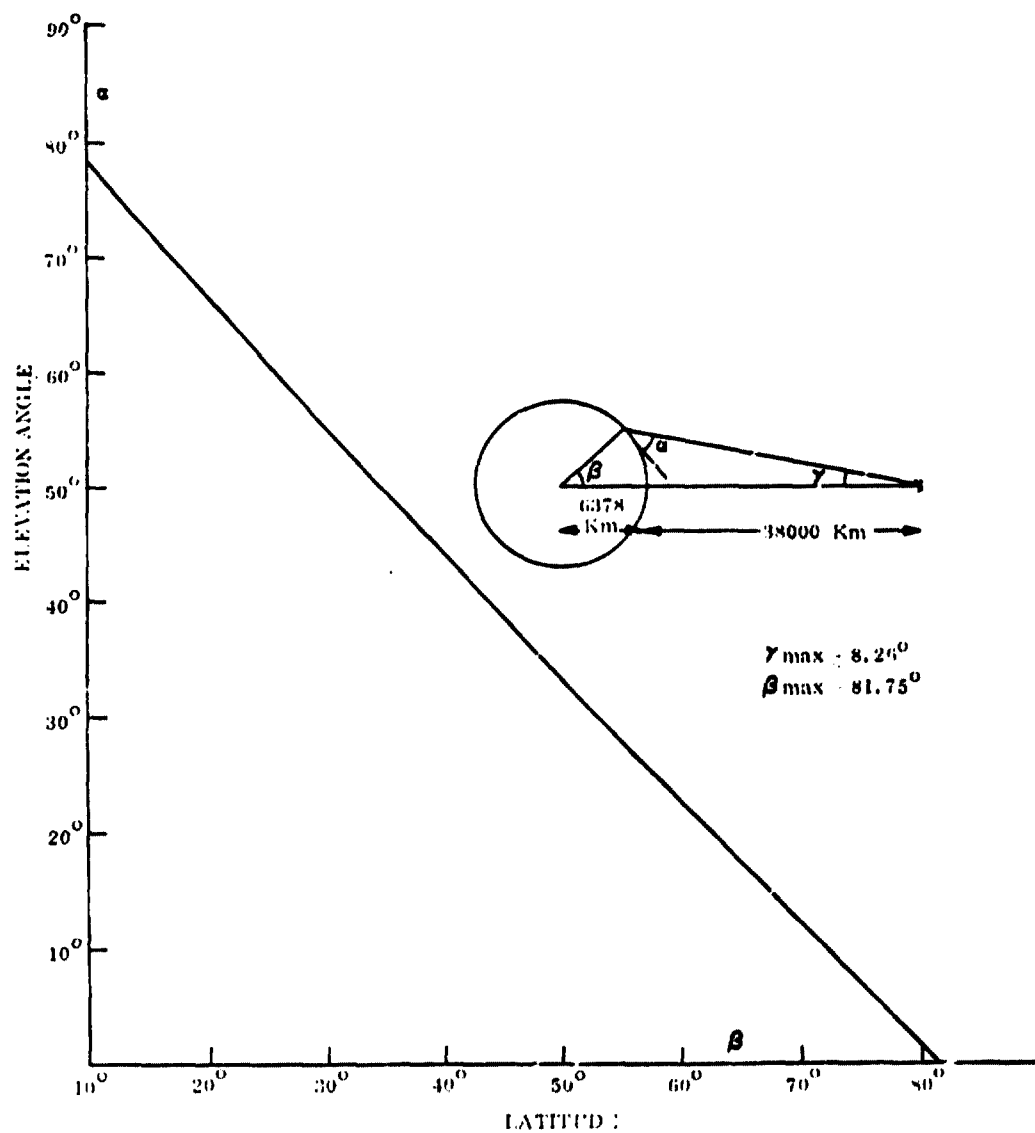
ELEVATION ANGLES FOR SPS AT DIFFERENT LATITUDES

IT IS NOT RECOMMENDED USING ELEVATION ANGLES BELOW  $20^{\circ}$ , WHICH CORRESPOND TO LATITUDE LOCATIONS ABOVE  $62^{\circ}$ .

LOW ELEVATION ANGLES ARE SUSCEPTIBLE TO DUCT EFFECTS AND MORE DEGRADATION FOR THE MAIN BEAM AND THE SIDELobe ENVELOPE.



## ELEVATION ANGLES FOR SPS AT DIFFERENT LATITUDES

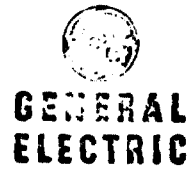


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**POWER DENSITY OVER RECTENNA**

THE FORMULA SHOWS RELATIONSHIPS BETWEEN GEOMETRY AND POWER DISTRIBUTION



## POWER DENSITY OVER RECTENNA



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- RADIAL DISTANCE RM ( IN KM ) AT ANGLE  $\Phi$

$$RM = \frac{B}{\sqrt{\sin^2 \Phi + \cos^2 \Phi \sin^2 \alpha}} = \frac{B}{SK}$$

$\alpha$  = ELEVATION BEAM ANGLE

B = SEMI MINOR AXIS DISTANCE

- POWER DENSITY

$$P_{RN} = P_0 \cdot e^{-k \times SK^2 \times RN^2}$$

$$P_0 = 21 \text{ mw/cm}^2$$

$$k = 0.10364$$

RN = ANY RADIAL DISTANCE OF RECTENNA

RECTENNA LAYOUT

THE INCLINATION ANGLE OF THE MAJOR AXIS OF THE RECTENNA IS DETERMINED BY THE LONGITUDE LOCATION OF THE RECEIVING SITE.

ALL THE PANELS OF THE RECTENNA, OF ELLIPTIC RINGS, OF EQUAL POWER LEVELS, ARE ARRAYED IN PARALLEL.

PRESENT DIVISION OF POWER USES 100 RINGS, EACH DELIVERS 50 MW OF D.C. POWER. EACH RING IS DIVIDED INTO 50 1 MW PRIMARY UNITS.

THE NUMBER OF RADIATION DETECTION ELEMENTS IS 15.7 BILLION.

POWER FLOW IS RADIAL FROM THE CENTER OF THE RECTENNA.

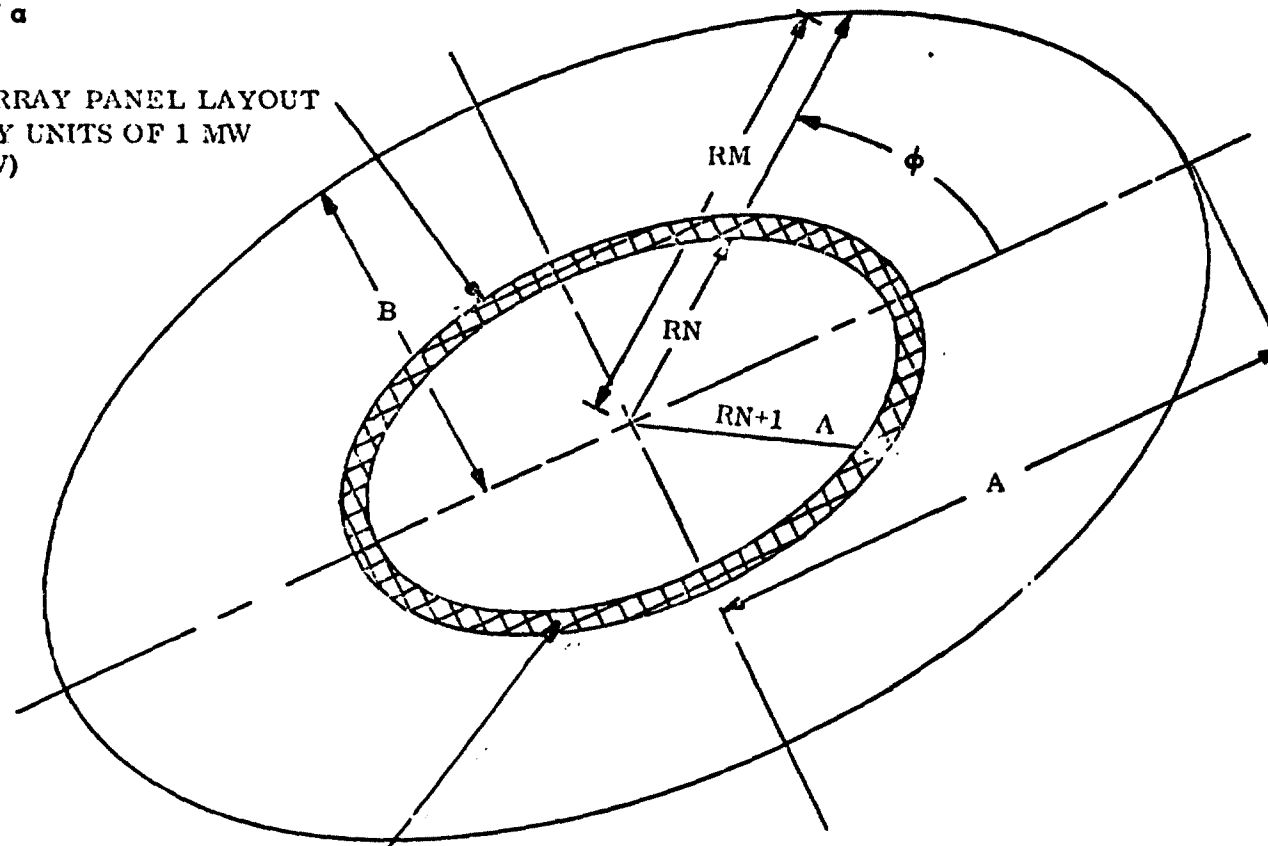


## RECTENNA LAYOUT



GROUND POWER OUTPUT = 5GW  
 MINOR AXIS (2B) = 10 KM  
 NUMBER OF RINGS (M) = 100  
 $B = A \sin \alpha$

50 MW ARRAY PANEL LAYOUT  
 (PRIMARY UNITS OF 1 MW  
 AT  $\pm 2$  KV)

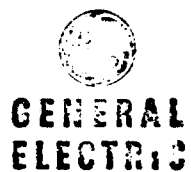


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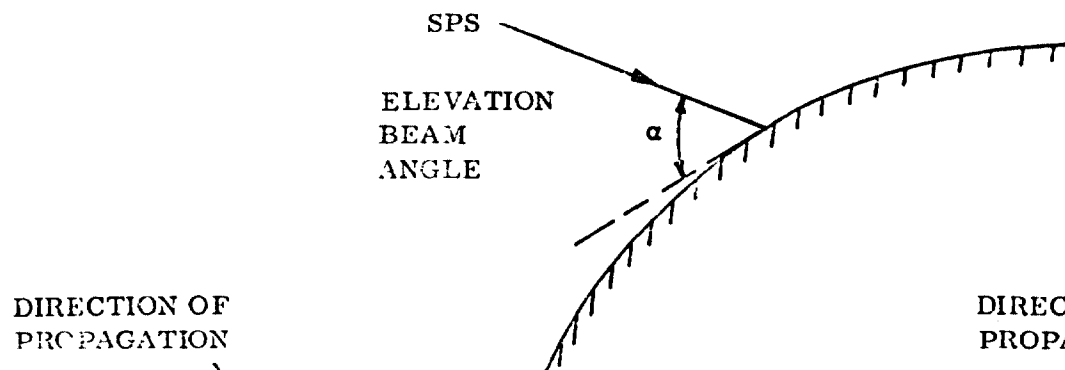
ELLIPTIC RINGS OF EQUAL  
 POWER LEVELS

LAYOUT OF RECTENNA PANELS

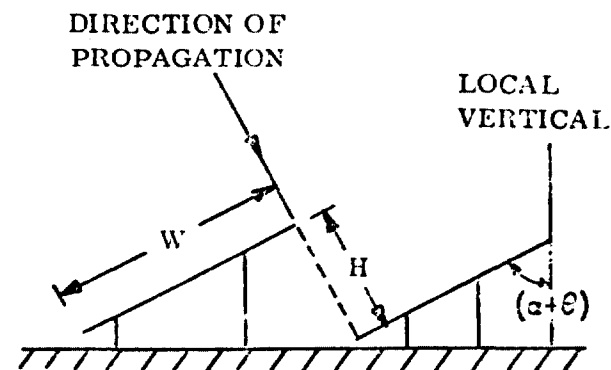
- o THE RECTENNA STRUCTURE IS DETERMINED BY THE TERRAIN CONFIGURATION, THE DIFFRACTION LIMITATIONS OF THE PANELS AND THE LATITUDE-LONGITUDE LOCATION.
- o THE CHOICE OF THE NUMBER OF RADIATING ELEMENTS ON EACH PANEL IS A FUNCTION OF THE DIRECTIVITY OF THE RADIATING ELEMENTS, THE ADDITIONAL ANGLE WHICH IS NEEDED TO YIELD A REASONABLE VALUE FOR K AND THE LOSSES OF POWER CARRYING WIRES BETWEEN LARGE SEPARATED ELEMENTS AT HIGH LATITUDES OF RECTENNA LOCATIONS.
- o THE RELATION BETWEEN THE ELEVATION BEAM ANGLE AND THE LATITUDE OF RECTENNA LOCATION ( OF SPS LONGITUDE) IS SHOWN IN THE FOLLOWING CHART.



## LAYOUT OF RECTENNA PANELS



UNLEVELED RECTENNA



LEVELED RECTENNA

$$H < K^2 N^2 \lambda$$

$$W = N \lambda$$

$\lambda$  = WAVE LENGTH  
 $K$  = SHADOWING FACTOR  
 (SMALL FRACTION)



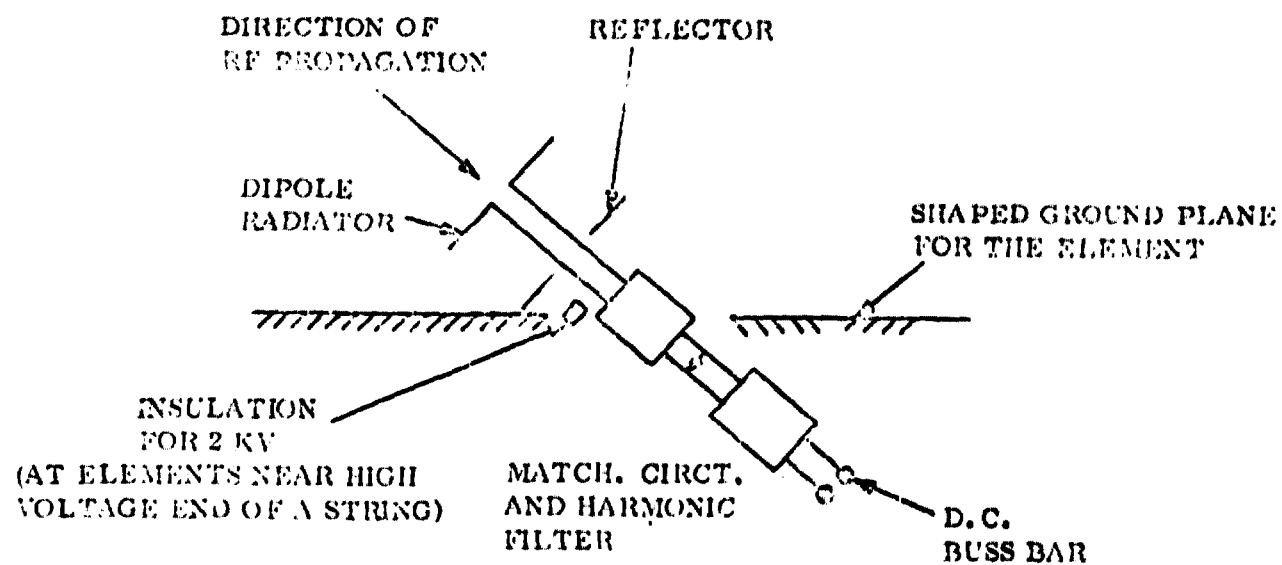
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PREFERRED POLARIZATION ATTITUDE

THE DIFFRACTION LIMITATIONS OF THE RECTENNA PANELS PROHIBIT USING A GROUND PLANE ORTHOGONAL TO THE WAVE FRONT AT HIGH LATITUDES. THIS NECESSITATES USING ADDITIONAL REFLECTOR TO THE DIPOLE RADIATORS. THIS FUNCTIONS PROPERLY WHEN THE POLARIZATION IS PARALLEL TO THE MAJOR AXIS OF THE RECTENNA.



# PREFERRED POLARIZATION ATTITUDE



PREFERRED POLARIZATION IS PARALLEL TO THE MAJOR AXIS OF ELLIPTIC SHAPED RECTENNA

RECTENNA EFFICIENCY

POWER COLLECTION EFFICIENCY INCREASES BY INCREASED ELEVATION DUE TO IMPROVEMENTS IN SEQUENTIAL CHARGING, INCREASED CURRENTS IN PARALLEL ARRAYS OF ELEMENTS AND LARGER PRIMARY UNITS.

D.C. TRANSMISSION EFFICIENCY FROM PRIMARY UNITS ELEVATION DECREASES WITH LATITUDE AND INCREASED ANGLE.



RECTENNA EFFICIENCY (RF/DC)  
(5.839 GW/5 GW)



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|                                                                |     |
|----------------------------------------------------------------|-----|
| D.C. CONVERSION EFFICIENCY OF ELEMENTS                         | 89% |
| POWER COLLECTION EFFICIENCY                                    | 98% |
| D.C. TRANSMISSION EFFICIENCY                                   | 97% |
| (99.5% AT CENTER & 94% AT PERIPHERY)                           | —   |
| TOTAL D.C. CONVERSION EFFICIENCY<br>(D.C. POWER OUTPUT = 5 GW) | 85% |

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**RECTENNA COST**

COST OF DIPOLE ASSEMBLY IS BASED ON ASSUMPTION THAT \$0.1 WOULD BE REASONABLE FOR MASS PRODUCTION OF  $15,710 \times 10^6$  ELEMENTS NEEDED FOR ONE RECTENNA. PROJECTED COST OF SCHOTTKY BARRIER DIODES 1¢/ DIODE. SUPPORT STRUCTURE AND LAND IS ESTIMATED ON THE BASIS OF SYSTEM #7 OF BOVAY INGRS., INC. REPORT OF JUNE 20, 1977. (GROUND PLANE @ \$.36/LB.)

BUS BAR COST IS ESTIMATED ON THE BASIS OF ROCKWELL'S EVALUATION.



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RECTENNA COST



|                             |                                 |
|-----------------------------|---------------------------------|
| DIPOLE ASSEMBLY             | $\$ 1,571 \times 10^6$          |
| IMPROVED S.B. DIODES        | $\$ 157 \times 10^6$            |
| POWER DISTRIBUTION BUS BARS | $\$ 57 \times 10^6$             |
| SUPPORT STRUCTURE           | $\$ 2,051 \times 10^6$          |
| LAND (TYPICAL MIDWEST)      | $\$ \underline{71} \times 10^6$ |
| TOTAL RECTENNA              | $\$ 3,907 \times 10^6$          |

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REQUIRED INVESTIGATIONS

IMPROVED RADIATING ELEMENTS WHICH MAY BE MASS PRODUCED ECONOMICALLY (ESPECIALLY IF THE SPS IS STABLE ENOUGH TO ALLOW USING SMALLER NUMBER OF HIGHER GAIN ELEMENTS).

ANALYSIS OF POWER FLOW SCHEMES WHICH ACCURATELY ASSESS THE D.C. COLLECTION EFFECIENCY BASED ON THE MANUFACTURING TOLERANCES OF DETECTION ELEMENTS AND THE SLOW OSCILLATORY IONOSPHERIC AND ATMOSPHERIC EFFECTS.

EVALUATION OF RE-RADIATION OF CROW-BARED RECTENNA OF THE RANDOM PHASE FRONT CONCERNING RADIATION HAZARD.

EVALUATION OF GRATING LOBES OF SCATTERED RADIATION OF PERIODIC RECTENNA STRUCTURES.

EFFECT OF HEATED IONOSPHERE ON SCINTILLATION LEVELS OF THE RECTENNA AND ON SCATTERED RADIATION LEVELS.



REQUIRED INVESTIGATIONS



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GROUND POWER COLLECTION  
AND  
TRANSMISSION SYSTEM

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KEY CHARACTERISTICS OF SPS-UTILITY INTERFACE

THE CHARACTERISTICS OF THE SPS WITH THE MOST IMPACT ON THE UTILITY SYSTEM IN AN OPERATIONAL SENSE IS THE LARGE BLOCK OF POWER CONCENTRATED IN ONE UNIT, AND THE RELATIVE INFLEXIBLE NATURE OF A CONSTANT ENERGY SOURCE.

THE SPS CHARACTERISTICS IN LIGHT OF THE ELECTRIC UTILITY SERVICE CRITERIA OF ADEQUATE SYSTEM RELIABILITY AND MINIMIZING COST TO THEIR CUSTOMERS REQUIRES SPECIAL ATTENTION WHEN DESIGNING THE GROUND POWER COLLECTION AND TRANSMISSION SYSTEM BETWEEN THE REFERENCE PRIMARY UNITS AND THE CONNECTION/INSERTION POINT OF THE UTILITY BULK POWER GRID.

THE UTILITY LOAD CHARACTERISTICS HAVE PARTICULAR IMPACT, SINCE THE SPS GROUND SYSTEM HAS NO AUTOMATIC LOAD FOLLOWING CAPABILITY AND OTHER GENERATION CAPACITY WILL BE NEEDED AT ALL TIMES TO PROVIDE FOR UTILITY SYSTEM FREQUENCY CONTROL AND SPINNING RESERVE NEEDS.

**KEY CHARACTERISTICS OF SPS-UTILITY INTERFACE**

**SPS:**

- **LARGE UNIT SIZE**
- **CONSTANT ENERGY SOURCE**

**UTILITY:**

- **SERVICE QUALITY CRITERIA**
  - **RELIABILITY**
  - **MINIMIZING COST TO CUSTOMERS**
- **VARIABLE LOAD DEMAND**
  - **BY HOUR**
  - **BY SEASON**
  - **BY YEAR**

DESIGN CONSIDERATIONS FOR RECTENNA GROUND  
POWER COLLECTION AND TRANSMISSION SYSTEM

TO ASSURE RELIABILITY IT WAS AN OBVIOUS NEED TO USE A MODULAR CONCEPT, INCORPORATING EQUIPMENT OF KNOWN TECHNOLOGY AND PERFORMANCE. THE DESIGN PRINCIPLES USED FOLLOWS CONVENTIONAL UTILITY DESIGN PRACTICE FOR CONTROL AND PROTECTION.

TO MINIMIZE COST THE PLANT LAYOUT FOLLOWS A SYMMETRICAL CONCEPT AND EQUIPMENT RATINGS LIKE CONVERTER STATION POWER LEVELS AND SYSTEM VOLTAGE LEVELS ARE CHOSEN TO BE COMPATIBLE WITH RECTENNA PRIMARY UNITS, AND MINIMIZE COST WHILE PROVIDING MAINTAINABILITY.

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DESIGN CONSIDERATIONS FOR RECTENNA GROUND  
POWER COLLECTION AND TRANSMISSION SYSTEM

● RELIABILITY

- MODULAR CHARACTER
- EQUIPMENT PERFORMANCE
- CONTROL AND PROTECTION

● COST

- PLANT LAYOUT
- EQUIPMENT, STRUCTURES AND MATERIAL
- OPERATION AND MAINTENANCE

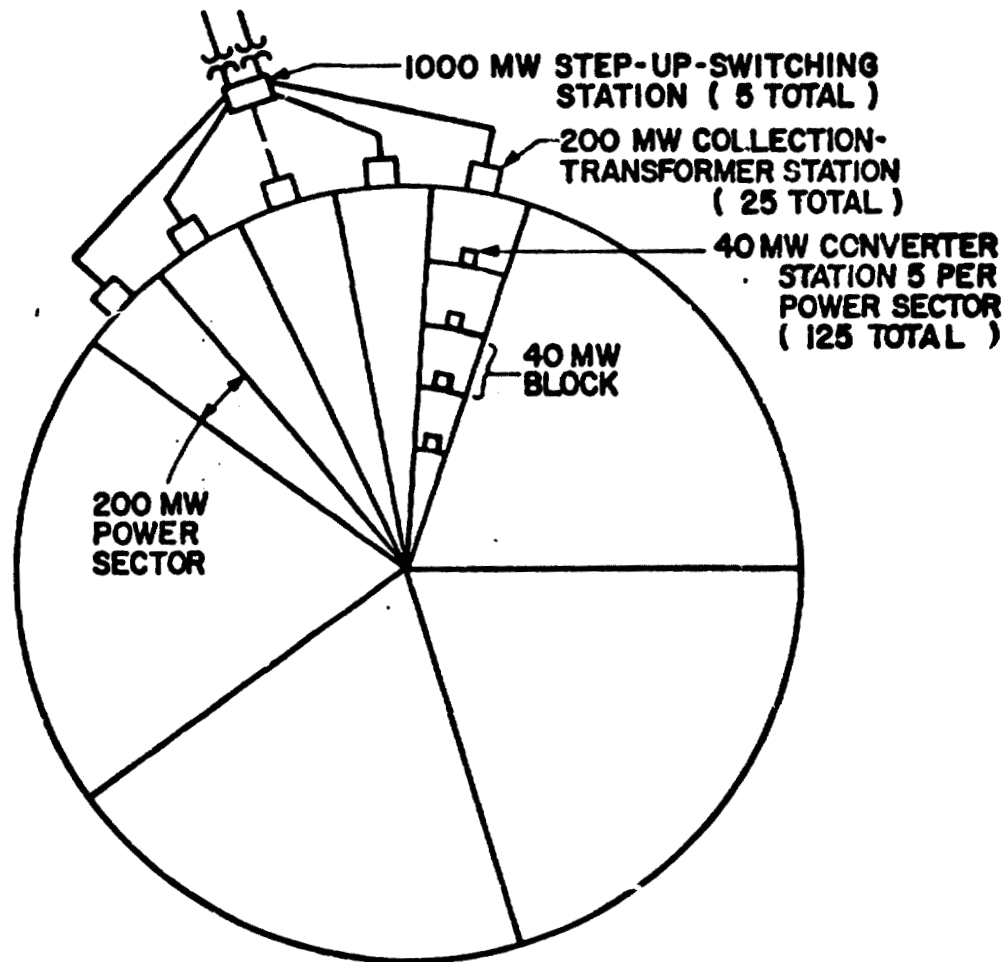
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GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

GENERAL LAYOUT

THE PLOT PLAN IS ASSUMED TO BE CIRCULAR WITH A TOTAL NET OUTPUT OF 5000 MW. THE RECTENNA AREA IS DIVIDED INTO 5 EQUAL AREAS EACH FEEDING ONE 1000 MW STEP-UP-SWITCHING STATION. EACH STEP-UP-SWITCHING STATION IS IN TURN FED BY FIVE 200 MW POWER SECTORS. EACH POWER SECTOR CONTAINS FIVE 40 MW BLOCKS. EACH CONVERTER STATION COLLECTS 40 MW DC POWER FROM PRIMARY RECTENNA UNITS AND INVERTS DC TO AC POWER.

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**RECTENNA POWER COLLECTION  
AND TRANSMISSION SYSTEM**



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GROUND POWER COLLECTION AND CONVERSION SYSTEM

EACH 40 MW POWER BLOCK CONSISTS OF FORTY 1 MW PRIMARY UNITS WITH OUTPUT VOLTAGE OF  $\pm 2$  kV. THE END OF EACH PRIMARY UNIT IS CONNECTED THROUGH DC CIRCUIT BREAKERS TO 2 kV DC CABLES RUNNING RADIALY AS SHOWN IN THE DIAGRAM TO THE CONVERTER STATION. DC SMOOTHING REACTORS REDUCE THE RIPPLE CURRENTS.

SINCE THE RECTENNAS ARE CONSTANT POWER DEVICES AND THE DC/AC CONVERTER CAN IN NO WAY AFFECT POWER FLOW, THE CONTROL OF POWER BE APPLIED ON THE DC SIDE. THIS MEANS THAT EITHER THE RF LEVEL MUST BE CONTROLLED AT ITS SOURCE OR THE NUMBER OF RECTENNAS CONNECTED IN PARALLEL MUST BE VARIED. CIRCUIT BREAKERS PROVIDED FOR RECTENNA PROTECTION CAN ALSO BE USED TO ADD OR REMOVE UNITS IN ORDER TO CONTROL POWER, BUT NOT ON A CONTINUOUS BASIS.

IT IS RECOGNIZED THAT THE SPS SYSTEM WILL OPERATE AT CONSTANT POWER BUT POWER VARIATION IS NEEDED TO GET ON LINE AND TO GET OFF LINE FOR SUCH THINGS AS MODIFICATION OR MAINTENANCE. IT MAY BE THAT THE 20 MW POWER GROUPS ARE SMALL ENOUGH THAT THEY CAN BE PICKED UP OR DROPPED AS THE MINIMUM SIZE INCREMENT.

THE CONVERTER THYRISTOR BRIDGE CIRCUIT FEEDS ALTERNATING CURRENT TO THE CONVERTER TRANSFORMER WHICH STEPS THE VOLTAGE UP TO 69 kV AT 60 HZ.

FILTERS CONNECTED THE AC BUS ABSORB CURRENT HARMONICS GENERATED IN THE CONVERTER. THE AC WAVE SHAPE IS THEREBY KEPT WITHIN ACCEPTABLE HARMONIC CONTENT LIMITS FOR THE UTILITY GRID AND ASSOCIATED PLANT EQUIPMENT.

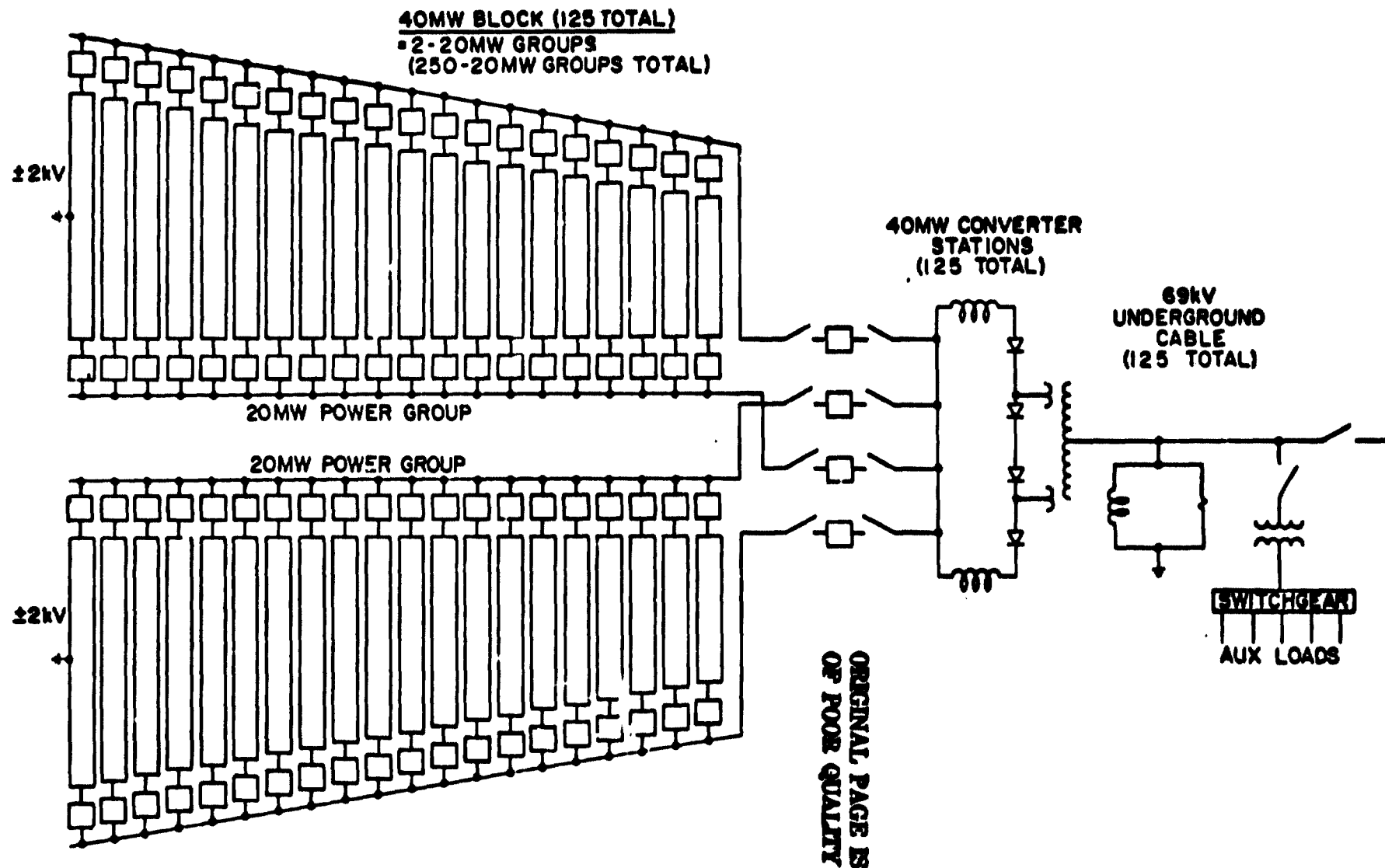
THE CONVERTER STATION OUTPUT, AT 69 kV AND A MAXIMUM CURRENT OF 400 AMPERES IS TRANSMITTED BY UNDERGROUND CABLE TO THE TRANSFORMER STATION.

THE CONVERTER STATION, ONCE COMMISSIONED, OPERATES AUTOMATICALLY. ALL SWITCHING, STARTUP AND SHUTDOWN ARE DIRECTED AND MONITORED BY A SMALL COMPUTER SYSTEM IN CONJUNCTION WITH OTHER CONVERTER AND STATION CONTROL EQUIPMENT.

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GROUND POWER COLLECTION AND CONVERSION SYSTEM



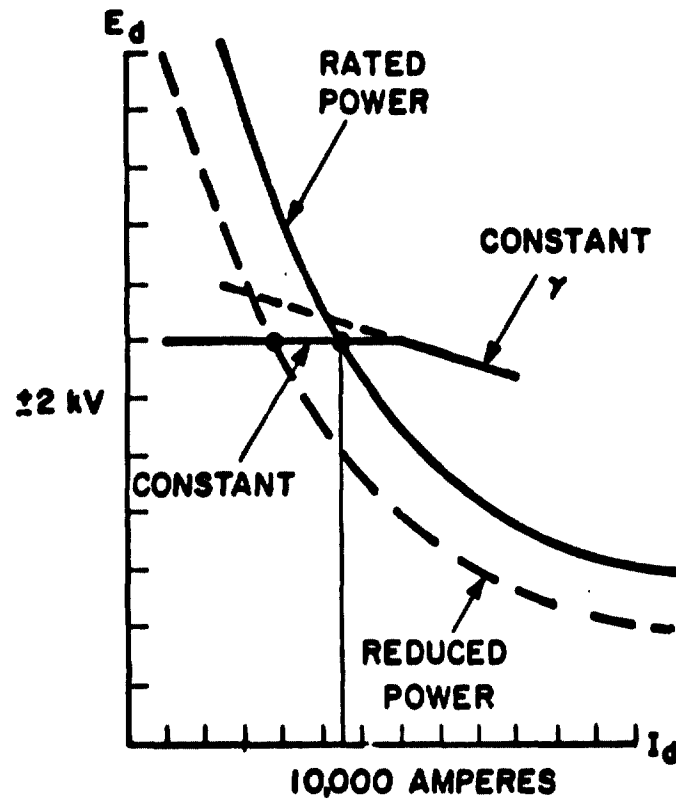
CONVERTER CHARACTERISTICS AND CONTROL BLOCK DIAGRAM

1. CHARACTERISTIC CURVE OF THE RECTENNA - THE VOLTAGE-CURRENT CHARACTERISTIC OF THE RECTENNA, OVER THE RANGE TO BE CONSIDERED, CAN BE DESCRIBED AS A CONSTANT POWER RECTANGULAR HYPERBOLA. AT HIGH VOLTAGE AND LOW CURRENT, AN AUTOMATIC SHORT CIRCUITING DEVICE OR "CROWBAR" WILL BE PROVIDED AS AN INTEGRAL PART OF THE RECTENNA. LIKEWISE AT THE LOW VOLTAGE, HIGH CURRENT END, THE CHARACTERISTICS WILL PROBABLY BE TERMINATED BY A SHORT CIRCUIT.
2. CHARACTERISTIC CURVE OF THE POWER CONDITIONING SYSTEM - THE INVERTER OF THE POWER CONDITIONING SYSTEM IS AN ELECTRONIC DEVICE CAPABLE OF SEVERAL MODES OF CONTROL. THE LINE-COMMUTATED INVERTER IS CHOSEN FOR THE SPS SYSTEM AND THE INVERTER IS OPERATED IN A CONSTANT VOLTAGE MODE.
3. CONTROL BLOCK DIAGRAM - THE "REGULATOR" IS SHOWN FEEDING A TRANSDUCER T WHICH CONVERTS A VOLTAGE INTO AN ANGLE FOR THE FIRING PULSES TO THE THYRISTORS OF THE CONVERTER BRIDGE CIRCUIT.

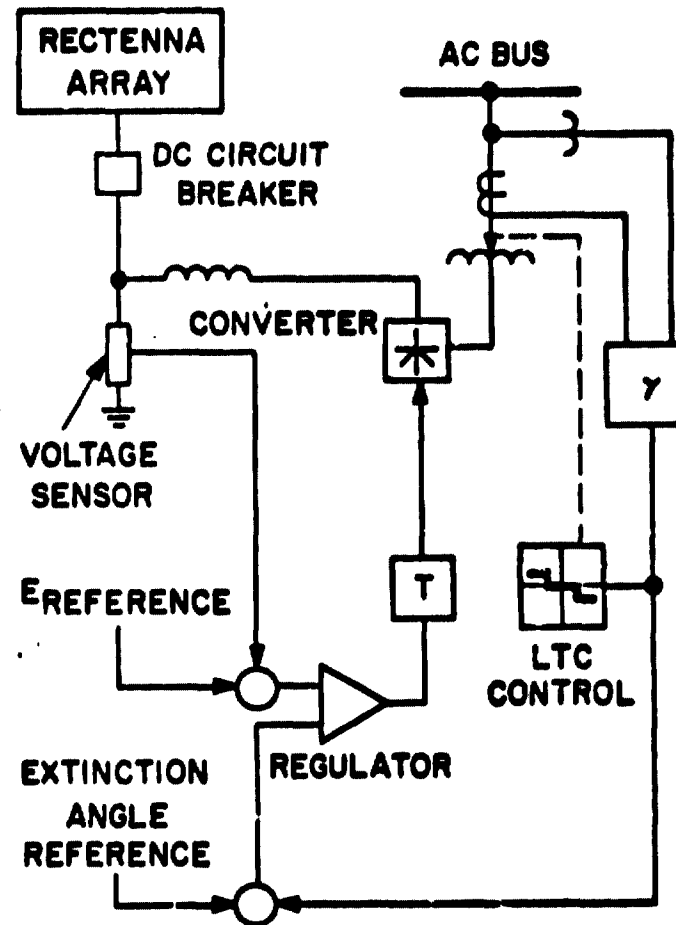
A DC VOLTAGE SENSOR PROVIDES A MEASURE OF THE ACTUAL VOLTAGE WHICH PROVIDES THE ERROR SIGNAL FOR THE REGULATOR. THIS AUTOMATIC VOLTAGE CONTROL LOOP IS THE PRIMARY CONTROL OF THE CONVERTER.

AC LINE CURRENT AND AC LINE VOLTAGE ARE MEASURED (AS SHOWN) AND A RELATIVE TIMING SIGNAL IS DEVELOPED WHICH RESULTS IN MEASUREMENT OF EXTINCTION ANGLE  $\gamma$ . THIS QUANTITY IS COMPARED TO THE EXTINCTION ANGLE REFERENCE AND THE RESULTING ERROR SIGNAL PASSES TO THE REGULATOR AND HOLDS THE CONSTANT  $\gamma$  CURVE WHEN CONDITIONS ARE SUCH THAT CONSTANT VOLTAGE CANNOT BE HELD.

### CONVERTER CHARACTERISTIC



### CONVERTER CONTROL BLOCK DIAGRAM



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**GROUND POWER COLLECTION AND TRANSMISSION SYSTEM**

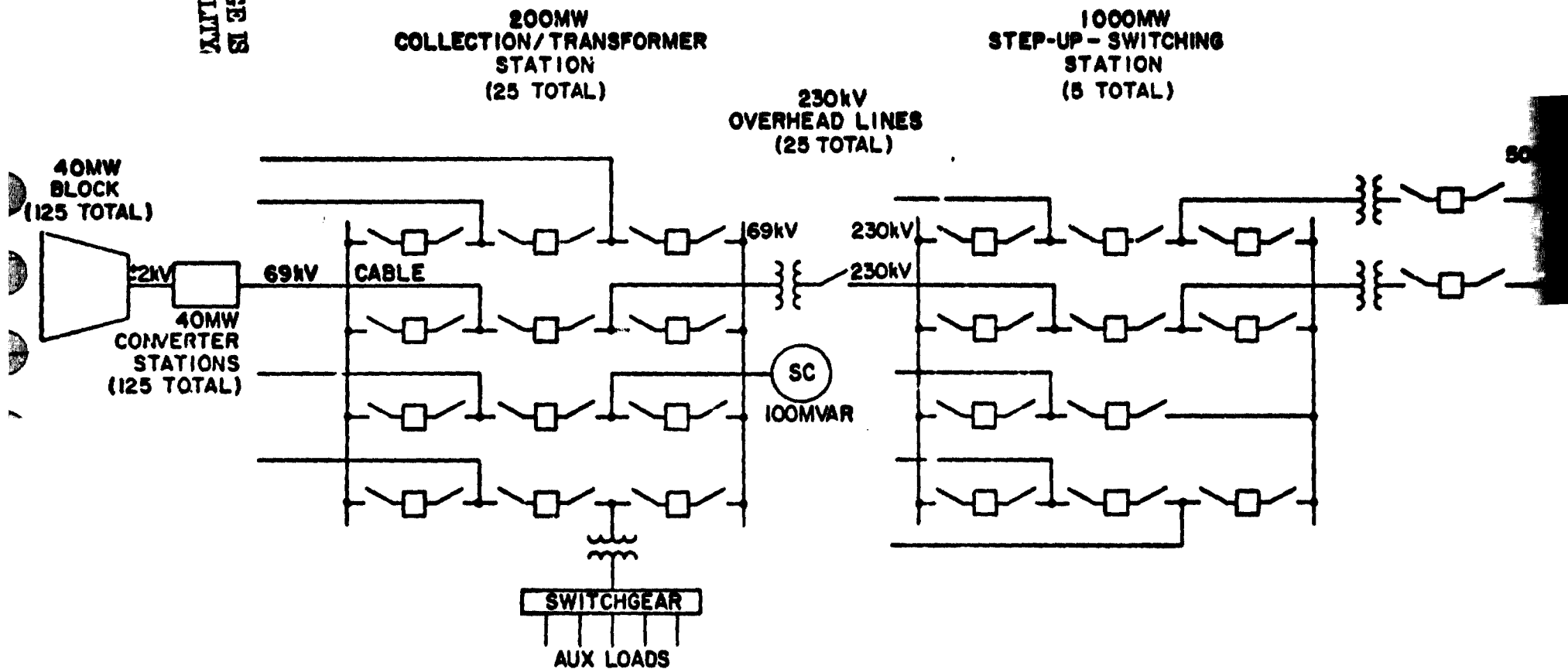
**ONE LINE DIAGRAM**

THE COLLECTION/TRANSFORMER STATION GATHERS THE POWER OUTPUT OF 5 CONVERTER STATIONS, CONNECTS THESE CIRCUITS INTO A RELIABLE SWITCHING ARRANGEMENT, AND TRANSFORMS THE AC POWER FROM 69 kV UP TO 230 kV. THIS IS DONE BY PHYSICALLY AND ELECTRICALLY ARRANGING AND CONNECTING STANDARD ELECTRICAL EQUIPMENT INTO THE DESIRED CONFIGURATION. THE ELECTRICAL CONFIGURATION PROVIDES RELIABILITY BY A "BREAKER AND A HALF" SCHEME 69 kV SWITCHYARD. A SINGLE CONTINGENCY OUTAGE CAN BE SUSTAINED IN THE 69 kV SWITCHYARD WITHOUT LOSS OF POWER OUTPUT CAPABILITY. TO PROVIDE COMPENSATION FOR THE INHERENT LAGGING POWER FACTOR CHARACTERISTICS OF THE CONVERTER VALVE AND TRANSFORMER EQUIPMENT ONE 100 MVAR SYNCHRONOUS CONDENSER IS CONNECTED TO THE 69 kV BUS. THE SYNCHRONOUS CONDENSER RATING IS CHOSEN TO ALLOW SYNCHRONOUS CONDENSER MAINTENANCE ON ADJACENT COLLECTION/TRANSFORMER STATIONS WITHOUT CURTAILING POWER OUTPUT.

THE STEP-UP SWITCHING STATION RECEIVES THE OUTPUT FROM FIVE COLLECTION/TRANSFORMER STATIONS AT 230 kV AND TRANSFORMS THE VOLTAGE TO 500 kV. THE "BREAKER AND A HALF" SCHEME EMPLOYED CAN SUSTAIN ANY SINGLE CONTINGENCY 500 kV SWITCHYARD FAULT WITHOUT REDUCTION IN STATION OUTPUT. THE SELECTION OF THE VOLTAGE LEVEL FOR THE ULTIMATE BULK POWER TRANSMISSION INTERFACE WITH THE UTILITY GRID AS WELL AS THE POSSIBILITY OF INTERCONNECTING TWO OR MORE OF THE 1000 MW SWITCHING STATIONS TOGETHER SHOULD BE OPTIMIZED BASED ON DETAILED INFORMATION ABOUT THE CONNECTING UTILITY SYSTEM. THE SOLUTION SHOWN IS ONE OF SEVERAL POSSIBLE.

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# GROUND POWER COLLECTION AND TRANSMISSION SYSTEM



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GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

COST ESTIMATES

THE COST ESTIMATES ARE BASED ON THE CONCEPTUAL DESIGN SHOWN ON THE ONE-LINE DIAGRAM. THE COST INCLUDES MAJOR EQUIPMENT, BUILDINGS, MATERIAL AND DIRECT LABOR.

NOT INCLUDED ARE:

- INDIRECT LABOR
- A/E FEES
- CONTINGENCY
- INTEREST DURING CONSTRUCTION
- INSURANCE
- TAXES
- TRANSPORTATION OF EQUIPMENT AND MATERIALS TO SITE

ALL COSTS ARE IN 1978 U.S. DOLLARS.

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GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

SUMMARY OF  
COST ESTIMATES OF MAJOR EQUIPMENT,  
BUILDINGS, MATERIAL & DIRECT LABOR

| <u>QUANTITY</u> | <u>ITEM</u>                    | <u>TOTAL COST (M\$)</u><br><u>(1976 DOLLARS)</u> |
|-----------------|--------------------------------|--------------------------------------------------|
| 1               | PLANT CONTROL CENTER           | 2.9                                              |
| 5               | STEP-UP-SWITCHING STATION      | 22.6                                             |
| 39 km           | 230 kV TRANSMISSION LINE       | 4.2                                              |
| 25              | COLLECTION-TRANSFORMER STATION | 72.0                                             |
| 25              | 100 MVAR SYNCHRONOUS CONDENSER | 87.5                                             |
| 381 km          | 69 kV CABLE & CONDUIT          | 69.5                                             |
| 125             | CONVERTER STATION              | 307.0                                            |
| 5443 km         | 2 kV DC CABLE/BUS              | 48.6                                             |
| 10,000          | 2 kV, 250 A DC CIRCUIT BREAKER | <u>60.0</u>                                      |
|                 | <b>TOTAL</b>                   | <b>674.3</b>                                     |

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GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

OPERATION AND MAINTENANCE

THE NORMAL MODE OF OPERATION WILL BE AS A CONSTANT ENERGY SOURCE. THE UTILITY SYSTEM HAS NEEDS FOR FLEXIBLE GENERATION FOR LOAD FOLLOWING, SPINNING RESERVE AND SYSTEM FREQUENCY CONTROL. THE SPS SYSTEM WILL NOT BE ABLE TO CONTRIBUTE TO FREQUENCY CONTROL AND SPINNING RESERVE, BUT COULD CONCEIVABLY GIVE NEED FOR ADDED GENERATION FOR SPINNING RESERVE AND FREQUENCY CONTROL. THE UTILITY OPERATIONAL CONSTRAINTS MIGHT WELL BE IMPORTANT CONSIDERATIONS IN DETERMINING THE PRACTICAL LIMITS TO THE PENETRATION OF SPS POWER IN FUTURE UTILITY SYSTEMS.

THE MODULAR DESIGN OF THE GROUND POWER COLLECTION AND TRANSMISSION SYSTEM SHOULD PROVIDE HIGH AVAILABILITY SINCE ROUTINE MAINTENANCE COULD BE PERFORMED ON A PARTIAL SHUTDOWN BASIS. CONSIDERABLE EFFECTIVENESS IN MAINTENANCE ROUTINES SHOULD ALSO BE EXPECTED ALTHOUGH A SIGNIFICANT SPARE PARTS INVENTORY WOULD APPEAR NECESSARY.



**GROUND POWER COLLECTION AND TRANSMISSION SYSTEM**

**OPERATION AND MAINTENANCE**

**NORMAL OPERATION:**

- **CONSTANT ENERGY SOURCE**
  - NO FREQUENCY CONTROL
  - NO LOAD FOLLOWING CAPABILITY EXCEPT BY SWITCHING MODULES
- **MAINTENANCE BY MODULAR SHUTDOWN**
  - HIGH AVAILABILITY
  - EFFECTIVE MAINTENANCE

GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

OPERATION AND MAINTENANCE

LOSS OF TOTAL OR PARTIAL POWER FROM THE SATELLITE WOULD OCCUR SEVERAL TIMES A YEAR DUE TO SATELLITE ROUTINE MAINTENANCE, SHADOWING EFFECT FROM THE EARTH AND FROM OTHER SPS SYSTEMS. WHEN THESE OUTAGES ARE PREDICTABLE AND SCHEDULED IT SHOULD HAVE A MINIMAL EFFECT ON THE UTILITY SYSTEMS INTEGRITY AND OPERATIONS SINCE THE TIMING OF SUCH OUTAGES WOULD BE DURING THE LOW LOAD PERIODS.

HOWEVER, ASSUMING A SIGNIFICANT PENETRATION OF SPS POWER SYSTEMS IN THE FUTURE, THE GENERATION RESERVE NEEDED TO MAINTAIN THE UTILITY SERVICE RELIABILITY WOULD BE EXPECTED TO INCREASE TO COVER THE EMERGENCY SHUTDOWNS OF THE SATELLITE. MORE DETAILED UTILITY SYSTEM STUDIES WOULD BE NEEDED TO PREDICT THE IMPACT ON RESERVE LEVELS FROM EMERGENCY SPS POWER SYSTEM OUTAGES.

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**GROUND POWER COLLECTION AND TRANSMISSION SYSTEM**

**OPERATION AND MAINTENANCE**

**LOSS OF POWER FROM SATELLITE:**

- **SCHEDULED MAINTENANCE**
  - **LOW IMPACT IN LOW LOAD PERIODS**
- **EMERGENCY SHUTDOWN**
  - **POTENTIAL NEED FOR INCREASED GENERATION RESERVE**

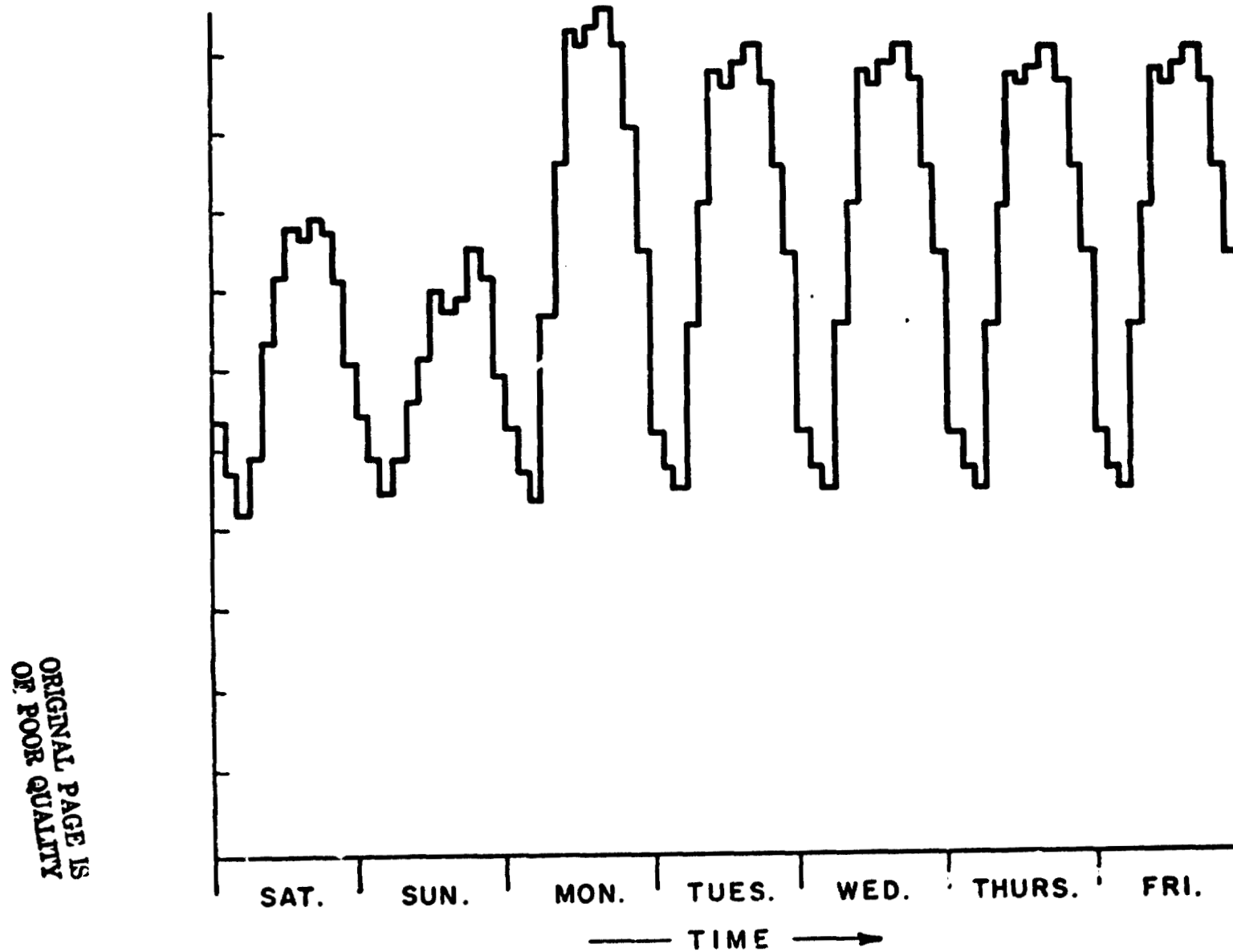
ELECTRIC UTILITY WEEKLY DEMAND PROFILE

THE ELECTRIC UTILITY DEMAND IS CHANGING BY THE SECOND, MINUTE, HOUR, DAY, SEASON AND YEAR. THE FIGURE SHOWS A TYPICAL ELECTRIC UTILITY WEEKLY DEMAND PROFILE.

THE GENERATION CAPACITY APPLIED TO SERVE THIS DEMAND MUST IN TOTAL BE CONTROLLABLE TO ADOPT TO THE DEMAND PROFILE. WITH A SIGNIFICANT AMOUNT OF INFLEXIBLE GENERATION INTRODUCED, THIS WILL CAUSE OPERATIONAL PROBLEMS IN THE LOW LOAD PERIODS. ONE RULE OF THUMB IS THAT AT LEAST 50% OF MINIMUM LOAD SHOULD BE CAPABLE OF CONTRIBUTING TO SYSTEM FREQUENCY CONTROL. THIS ASSUMPTION WOULD BE FOR A TOTALLY INTERCONNECTED SYSTEM WITH SEVERAL UTILITY POOLS CONNECTED TOGETHER. "SPINNING RESERVE" IS OPERATIONAL RESERVE CAPACITY PARTLY CONNECTED TO THE LINE AND PARTLY AVAILABLE WITHIN MINUTES. THIS RESERVE IS DESIGNED TO BE AVAILABLE IMMEDIATELY FOLLOWING EMERGENCY SHUTDOWNS OF OPERATING EQUIPMENT. IF THE UNIT SIZE OF A SPS POWER PLANT IS LARGE AS COMPARED TO OTHER LARGE UNITS ON THE SYSTEM, AN INCREASE IN THE SPINNING RESERVE REQUIREMENTS WOULD BE EXPECTED.

THE UNIT SIZE OF SPS POWER PLANT WOULD ALSO HAVE AN INCREASING EFFECT ON THE SYSTEM OVERALL GENERATION RESERVE FOR RELIABILITY. A FURTHER EVALUATION WOULD BE NEEDED INCLUDING UTILITY SYSTEM PARAMETERS, INDIVIDUAL UNIT RELIABILITY, AND SPS POWER SYSTEM OVERALL RELIABILITY AS INPUT DATA TO ASSESS THE MAGNITUDE OF THIS IMPACT.

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ELECTRIC UTILITY WEEKLY DEMAND PROFILE



GENERAL  ELECTRIC



GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

SUMMARY AND CONCLUSIONS

- DESIGN USING CONVENTIONAL TECHNOLOGY
- CONVENTIONAL UTILITY DESIGN PRACTICE
- HIGH RELIABILITY AND AVAILABILITY
- INSTALLED COST APPROX. 135 \$/KW

AREAS FOR FURTHER STUDY:

- SPS-UTILITY SYSTEM OPERATIONAL INTERFACE
  - RELIABILITY AND AVAILABILITY
  - OPERATIONAL ECONOMICS